QPR-3

Third Quarterly Progress Report
for
SPACE SYSTEMS ANALYSIS
and
COMPUTER PROGRAMMING SERVICES

1 January 1967 - 31 March 1967

Contract No. NAS 5-10022

Prepared by

Scientific Satellite Systems Department
Federal Systems Division
INTERNATIONAL BUSINESS MACHINES CORPORATION
Gaithersburg, Maryland

For

Goddard Space Flight Center
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Greenbelt, Maryland

18100 Frederick Pike
Gaithersburg, Maryland 20760
301 921-6000

April 15, 1967

National Aeronautics and Space
Administration
Goddard Space Flight Center
Greenbelt, Maryland

Attention: Mr. Robert J. Flick

Subject: Contract NAS 5-10022, Space Systems Analysis and Computer Programming Services

Gentlemen:

Submitted herewith is the quarterly progress report for the period from January 1, 1967, to March 31, 1967, as called for in the subject contract. It is the third quarterly progress report delivered under the contract.

Very truly yours,

Richard W. Gretz
Project Manager

RWG/btw

cc: Mr. J. J. Fleming
Mr. D. H. Gridley
Mr. A. G. Johnson
Dr. J. W. Siry

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ABSTRACT

This third quarterly progress report (QPR-3) describes the technical progress achieved on the Space Systems and Computer Programming Services Contract, NAS 5-10022, during the period January 1, 1967, through March 31, 1967. Work was performed on 21 separate tasks during the quarter in the areas of satellite attitude determination and steering, orbit determination, maneuver command, telemetry data processing and reduction and operations support. In addition work was performed on two tasks in the business management controls area.

Significant technical accomplishments during the quarter include support given to the launch and subsequent mission of three satellites (1) TIROS Operations Satellite-B, (2) Orbiting Solar Observatory-C, and (3) Applications Technology Satellite-A. Attitude determination and control was provided for the first two satellites. For the third, a computer program system was provided for processing PFM telemetry data. There were technical problems associated with the OSO-C support in that discrepancies in the attitude results and the prediction of attitude have occurred. At the time this report was published these problems have not been resolved. Support for the other two satellites was provided without incident.

Phase I of the attitude system redesign plan for the System/360 was completed (Phase I involved modification of 7094 programs to conform to rules providing minimum conversion effort). Phase II was carried from 50 to 63 percent completion.

In-orbit support of other than routine nature was provided for several satellites (1) Atmospheric Explorer-B, and (2) Applications Technology Satellite-B. For AE-B attitude determination was affected by large variations

in satellite spin rate and large scatter in instrument output. A significant analytical effort was required to keep track of attitude under these conditions. Several maneuvers were planned and carried out to adjust the orbit of the ATS-B satellite. A detailed analysis of the determination of attitude from picture data was performed to check attitude results.

A significant amount of definitive orbit work was performed on the ATS-B, GEOS, and POGO satellites. The ATS-B orbits were brought up to date on January 27, 1967. In the operational orbit area a standardized reporting system was evolved for the complete orbit data generation process employed and was placed in use.

The System/360 version of the GREMEX (Goddard Research Management Exercise) program was completed and delivered to NASA. Also the System/360 PAGE program for preparing project reporting charts was completed and delivered.

The conversion of the PERTAPE I program to System/360 was completed and the program tested.

The following recommendations and conclusions were made as a result of the work performed during the quarter.

A number of recommendations are made on the basis of the analysis of the support provided the December launch of ATS-B. They are detailed under Task 3. The most important of these recommendations deal with station coverage, limitations on firing at first apogee of the transfer orbit, and proper utilization of data (Task 3).

The constraint to maintain the axis of the Atmospheric Explorer-B satellite along the spin axis should be relaxed to permit the sun sensor to continue to view the sun. Also, the requirements for sun-only solution to attitude should be investigated because of poor magnetometer data (Task 6).

Thorough testing of the five-wire analog-to-digital system in conjunction with processing of the output should be performed prior to the TOS-C launch (Task 9).

Experimenters and their programming staffs should start working toward standardization of formats and methods. This move would make a generalized system much simpler and less costly (Task 16).

The computer program APMTR2 instead of APMTR should be used to determine conditions necessary for achieving an acceptable orbit for the Radio Astronomy Explorer Satellite (Task 19).

The experimenter's conference should be held at least four to six months prior to launch to allow time for data format and processing requests (Task 21).

Quality control personnel and programmers should work together daily as a team during the first month after launch to provide suitable interpretation of quality control parameters (Task 21).

The Definitive Orbit Determination System delivery date and scope of the Model 1 system should be reevaluated in view of increased systems scope and the lack of definitized system requirements (Task 25).

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Section I

INTRODUCTION

This report describes the technical progress achieved by the Scientific Satellite Systems Department, International Business Machines Corporation, on the Space Systems Analysis and Computer Programming Services Contract, NAS 5-10022, during the third quarter of the contract. The quarter includes the months of January through March 1967. The contract is of the task-order type involving on-site/off-site programming and analysis tasks in support of satellite orbit determination, satellite attitude determination, satellite scientific data processing and reduction, computer programming languages and executive systems and business data processing. The tasks include theoretical analysis studies and development, design, writing, testing, and documentation of computer program systems in support of scientific, communications, meteorological and related satellite projects at Goddard Space Flight Center (GSFC).

During the third quarter of the contract period three new tasks were assigned by Goddard Space Flight Center.

1. Task 25 - Develop a definitive orbit determination program system to run on the IBM System/360.
2. Task 26 - Develop an attitude determination and prediction system for support of the ESRO II Satellite.
3. Task 29 - Modify and convert the IBM 7010 PFM Telemetry Data Processing System to the UNIVAC 1108.

Task 10/23, GREMEX, and Task 22, PAGE, were completed during the quarter.

A brief summary of the work accomplished under each task is presented in the following paragraphs. In general, the assigned tasks involve satellite attitude determination and control, orbit determination, telemetry data processing, command generation, and maneuver control. Some business management systems programming is also included. Each summary gives a brief definition of the task to show the problem being addressed.

Technical reports delivered during the quarter are listed in Table 1.

Table 1

DOCUMENTS DELIVERED

Program Documentation of NIMBUS Ephemeris Programs	January 1967
Early Orbit and Attitude Determination Plan and Prelaunch Analysis for the Satellite TOS-B	January 1967
ESSA-4 Early Orbit and Attitude Determination Launch Day Report	January 1967
Users Guide to PERTAPE II	January 1967
Sync Finding Techniques	January 1967
Biosatellite-A Mission Summary	January 1967
Prelaunch Analysis and Attitude Determination System Description for the Satellite OSO-E1	February 1967
Final Specification for Processing of ATS-A EME PFM Telemetry Data Tapes (first revision)	February 1967
Analysis and Summary of ATS-B Apogee Motor Ignition, Maneuvers and Attitude Determination (draft form)	February 1967
Attitude Determination System for the AE-B Satellite (program documentation)	March 1967
Prelaunch Analysis and Attitude Control System Description for the Satellite OSO-E1	March 1967
OSO-E1 Early Orbit and Attitude Determination Launch Day Report	March 1967
Operating Instructions for OSO-E1 Attitude Determination System	March 1967
Final Specification for Processing of ATS-A EME PFM Telemetry Data Tapes (second revision)	March 1967

Task 1 - Biosatellite

Purpose: Complete the development of the BIOS Recovery Command Control System Program.

Summary: A study was made to determine methods of improving the efficiency of the BIOS-A launch version of the Recovery Control programs. As a result of the study, the criteria for selecting the time of retrofire has been relaxed and the integration steps during and after retrofire have been increased in some cases. These changes are being implemented and should improve the response time during the mission. Also, a method to reduce the near capacity core required by the three call-down routines has been successfully implemented and tested in two programs. The design of the System/360 version of the Recovery Control programs has been completed and conversion is well underway.

Task 2 - Documentation

Purpose: Provide technical services involving coordination of all technical reports, such as program documentation, and provide technical writing support. Also, study, develop and implement automatic techniques for program documentation.

Summary: Thirteen documents were completed and delivered to NASA. These documents include two volumes of program documentation, two prelaunch analysis reports, two launch day reports, one specification and two analytical reports. The camera copy (page impressions) for one of the volumes was made using an automatic (computer supported) documentation system. This system is being used to prepare the flowcharts for another documentation volume which will be delivered during the next quarter.

Task 3 - Applications Technology Satellite-B

Purpose: Develop computer software systems and analytical capability for support of the ATS-B mission in areas of spacecraft attitude determination and steering, and spacecraft maneuver and control.

Summary: A report summarizing an analysis of all major maneuvers of the ATS-B satellite following injection into the near synchronous orbit was prepared and delivered. This report describes the problems encountered during command of the December portion of the mission and details some recommendations for future missions. Formulation of an ATS-B attitude-determination technique involving least-squares fitting of ATS-acquired picture data was prepared and is currently being programmed. A preliminary error analysis of this technique and of in-orbit low thrust maneuvers for ATS were also undertaken during this quarter.

Task 5 - Attitude System Design

Purpose: Convert present operational computer programs used to support such projects as TOS/ESSA, NIMBUS, AE-B, OSO, and OGO to operation upon the IBM System/360.

Summary: Phase I of the Attitude System Design task as outlined in the report "Task Description of Initial System/360 Conversion Effort" dated 19 October 1966 has been completed. Phase I involved those System/360 compatible changes that it was possible to accomplish on the IBM 7094, and a general survey of Operating System/360 facilities and capabilities to determine their effect on the conversion effort.

Phase II, the actual conversion to operation on the System/360 Model 65, is well underway and is scheduled for completion next quarter. The conversion effort has concentrated on programs in the TOS/ESSA Attitude Determination System, Prelaunch and Utility Programs (PUP) and the Multi-Application Subroutines (MAS). The table below summarizes the status of the conversion effort.

	No. of Routines	Percent
Total no. of programs and subroutines being converted	36	100
Programs completed Phase I	36	100
Programs completed Phase II	23	64

Task 6 - Atmospheric Explorer-B

Purpose: Provide analysis, programming and operations support for post-launch satellite attitude determination, spacecraft control and command, and final attitude evaluation.

Summary: A study was made of the attitude data processing for the case of the AE-B spacecraft spinning in the opposite direction. Changes were made to the computer program system to account for this effect. In addition, a subroutine TIMSPN was programmed to compute the spin rate from midscan times with a correction made for the orbital angular velocity. Also, recent difficulties encountered in obtaining consistent AE-B attitude solutions from magnetometer and sun data have prompted an examination of the quality of the magnetometer data currently being processed.

Task 7 - NIMBUS Operations

Purpose: Provide analysis and operations for post launch orbit, attitude and picture prediction.

Summary: The operational support for the NIMBUS project is on schedule. This support included supplying World Map and Station Acquisition Data on a weekly and monthly basis, and on a definitive basis as requested.

Task 8 - Orbiting Solar Observatory

Purpose: Design and implement computer program systems for determination and prediction of attitude and attitude commands and provide operational support for attitude control.

Summary: The OSO-E1 launch and subsequent mission were supported in the area of attitude determination and prediction. Several new methods of determining spin rate and attitude were developed to provide backup.

Task 9 - TOS/ESSA

Purpose: Provide analysis, programming and operations support for prelaunch mission studies, launch window computations, satellite attitude determination, spacecraft control, and postlaunch analysis for the TOS/ESSA and TIROS series of satellites.

Summary: Excellent performance by the analysis, programming, and operational personnel involved in this task contributed to the successful attitude support of ESSA-4, launched 26 January 1967. A false bit in the Digital Sun Aspect Indicator data from the spacecraft was detected promptly by the satellite attitude operations group. Operational coverage of TIROS IX was resumed as ESSA-4 was turned over to the Environmental Sciences Services Administration after its two week checkout. A total of 629 messages from five satellites were processed through the ADS (Attitude Determination System).

Task 10/23 - Goddard Research Engineering Management Exercise

Purpose: Provide computer services in support of the GREMEX computer program. Convert the program to operate on the IBM 360/65 computer.

Summary: Conversion of the IBM 7094 GREMEX system to the System/360 was completed. Decks, listings, and documentation drafts were delivered to NASA on 2 February 1967.

Task 11/20 - Definitive Orbit Determination Analysis

Purpose: Develop operational and analytical capability for definitive orbit determination.

Summary: Emphasis centered on POGO and ATS-1 during the majority of this quarter as highest priority was given these satellites. Two-hundred POGO arcs were completed through the differential correction and error analysis stages. Orbital ephemeris data from 21 November 1965 to 1 March 1966 was delivered to the experimenter. Responsibility for ATS-1 orbit determination was assumed and work was brought up to date by 1 February 1967. Since then, results have been generated and delivered weekly. Two-hundred and seventy GEOS arcs were converged using all data types, completing this phase through 1 August 1966.

Task 13 - Launch Window

Purpose: Determine those intervals of time within which the IMP F satellite can be launched such that all constraints on the orbit and attitude of the satellite will be satisfied.

Summary: Preliminary results of the launch window study indicated several unsatisfactory conditions. A decision was reached by the IMP Project Office to conduct an exhaustive launch window study for IMP F for the restricted period from 17 May to 31 May 1967. For this purpose the Launch Window Program has been utilized to determine initial approximations of one year lifetime lines for nominal and plus 3 sigma injection parameters.

Task 14 - Orbit Operations

Purpose: Provide operational support in areas of orbit determination and orbit predictions.

Summary: Regular operational support, on a weekly basis, was provided for 38 satellites in the generation of weekly predicts and orbital tapes. Each week 83 regular runs were required and in the completion of all other requirements, an average of 700 computer runs were made each month. Nominal, prelaunch data was generated for 15 satellites, as requested, and three launches were actively supported. Special emphasis was placed on the generation of refined data in March and through week-end projects this work was brought nearly up to date. A new status reporting scheme was developed.

Task 15 - Scientific Data Analysis/General

Purpose: Design and develop computer programs for processing and analyzing scientific data from S-3, IMP and OGO class satellites.

Summary: The Encyclopedia Update and Logbook Update programs were expanded in scope to include processing of S3-A data, as well as S3 data. A request for modification of the S3 Encyclopedia data microfilm tapes was received and implemented during two weeks prior to scheduled delivery to Stromberg-Carlson.

The EPE-D Data Processing System, delivered for production last quarter, was modified to reduce the number of computer runs, the number of tapes required, and to economize on microfilm tape regeneration necessitated by intermittent output tape drive malfunctions. The EPE-D Attitude Determination study addressed the problem of recovery of attitude after the spacecraft passes through perigee where it tumbles.

The EGO-A Ency program was modified to eliminate the need for a 1410 computer pass over the experimenter decom tapes for blocking.

Two POGO-II programs were completed. One of these produces a highly condensed System/360 compatible attitude-orbit tape.

A three program system was completed for Dr. Balasubrahmanyan to plot combined data from IMP-A, IMP-B, IMP-C, and OGO-A vs. time on a three hour basis. The SUMOGO program conversion to System/360 was completed and changes initiated to conserve computer core storage, optimize running time, and simplify program logic for readibility and ease of modification.

The Gerber Plot Package documentation was delivered and the deck placed on the IBSYS library tape on the DCS computer. This eliminates the need for programmers to handle the large subroutine card deck.

The Konradi plot program was completed. This will be used to produce 102 orbital plots of electron and proton flux as well as geomagnetic and solar-magnetospheric latitude, magnetic local time and the Z(RE) attitude. Production of this program included design of special pre-formatted plot paper to cover 28 hours of data in four even-hour sections.

Task 16 - Scientific Data Analysis/360

Purpose: Design, develop, and put into operation a generalized System/360 data processing system to handle scientific data for the IMP-F and G, and the OGO-E and F satellites.

Summary: The scope of GSDAS was significantly expanded in that the system is now to be designed to satisfy potential users in all branches of the Laboratory for Space Sciences as well as the National Space Science Data Center. Implementation will be on both System/360 and the IBM 7094. Emphasis was placed on the redesign of system concepts and features to meet the expanded service requirements.

The specifications for the IMP-F Cosmic Ray Data Processing System were written and presented to the experimenter. Coding of the read subroutine for Dr. William's IMP-F Data Processing System was completed.

Task 19 - PERTAPE

Purpose: Modify and improve the GSFC PERTAPE computer program.

Summary: Program testing of APMTR2 (RAE Apogee Motor Program) was completed. Analysis, coding, and unit testing of LNRORB (Lunar Orbit Program) was completed. The conversion of PERTAPE I to System/360 was completed, and final listings, decks, and documentation were delivered to the Theory and Analysis Office.

Task 21 - ATS Data Reduction

Purpose: Design and develop computer program systems for processing of PFM telemetry data for the ATS-A and ATS-B projects.

Summary: Analysis, coding, unit testing, and system testing was completed on all programs in the ATS-A Data Reduction System, and all programs are operational for launch. Programming support was provided for the prelaunch checkouts of the A/D Line and EME package.

Post launch support for ATS-B was continued. A "User's Guide" for the ATS-B Data Reduction System was published. Work was concluded on this task except for program maintenance.

Task 22 - PAGE

Purpose: Produce a computer program for use on the IBM 360/65 which extracts data from the NASA PERT C Program and prepares a plot tape for generating milestone charts on the SC 4020 display.

Summary: The conversion of the IBM 1410 PAGE program to System/360 was completed. Program decks and listings and documentation drafts were delivered to NASA in March 1967.

Task 24 - SDA Documentation

Purpose: Provide composition, illustration, and printing production for documentation of all work performed during the period 1 July 1966 to 30 June 1967 in support of Tasks 15 and 16.

Summary: One document was delivered to NASA. Preliminary outlines and delivery schedules were prepared to show the organization, content, and delivery of the SDA program documentation. The task of preparing the individual program writeups for printing was started.

Task 25 - Definitive Orbit Determination System

Purpose: To specify a definitive orbit determination computer program system which will operate on the IBM 360 computer, and to perform systems integration, unit testing, and systems testing.

Summary: The DODS task got underway on 13 February 1967 and all milestones were adjusted for this start date. Activity during the quarter was primarily to assist GSFC in development of system requirements and formulation of the System Operating Requirements, a prerequisite for any systems design activity by IBM. Parallel effort was begun on items not particularly sensitive to system requirements, such as documentation and programming standards, design change procedures, basic math scientific support subroutines, OS/360 systems environment constraints, and preliminary study of the scope of data sets and some of the orbit determination math processors. However, it was not possible to begin specific concrete systems design activity during this quarter, as originally scheduled.

Task 26 - ESRO II Attitude System

Purpose: Develop and program an attitude determination system to back up the system being written by the ESRO group and to provide operations support at launch and for two weeks afterwards.

Summary: An attitude determination system was designed, coded, debugged, unit and system tested for the ESRO II spacecraft. This system is now ready and operational for launch. It includes (1) a main program, ESRO, to process the data and determine the attitude, (2) an attitude prediction program, ESRAPP, which will also track the automatic magnetorquer, and (3) a manual method for back up by which data will be extracted from a printout of the data tape and punched on cards for input to the main program, ESRO.

Task 29 - ATS-B Conversion

Purpose: Redesign and reprogram all programs written for the current IBM 7010 ATS-B Data Reduction System to run on the UNIVAC 1108.

Summary: This is a new task which was approved in March. Analysis of PHASE0, COMVER, Sequence Count Print, and the Buffer Tape Print Programs was initiated.

Section II

TASK REPORTS

Task 1

BIOS

DISCUSSION

After the 14 December 1966 launch and successful three-day support of the first Biosatellite mission, a study was undertaken to determine methods of increasing the overall efficiency of the Recovery Control System. The following areas were investigated:

a. Addition of further useful output on each page of the call-down catalogs, such as the yaw error angles output at the time the sun angles are computed and the orbit numbers provided for each command pass

b. The effect of the size of the retrofire and post-retrofire integration steps on the accuracy of impact determination

c. Improvement of the response times for the various call-down catalog routines by relaxing the criteria for selecting the time of retrofire

d. Methods to reduce the near-capacity core required by the three call-down catalog routines

The study to determine methods of improving Recovery Control System efficiency and reduce the number of core locations used has been completed and about 75 percent of the changes have been implemented. The results of the analysis are described in the following paragraphs.

The addition of further output on the call-down catalogs has been abandoned. This was a result of an agreement with Mr. Robert Jackson, of Ames Research Center, that the catalog output format should not be changed between BIOS-A and BIOS-B missions.

Analysis of the effect of the size of the integration steps (both during and after retrofire) on the final computed landing point indicated that an increase in the integration steps can be made in most instances without losing accuracy. In choosing new values, the trajectories run have been divided into three classes:

Class A—trajectories whose impact point is output as the recovery location by programs NOMERL, IMMED, and FIXTIM

Class B—dispersion calculation trajectories

Class C—trajectories used in computing the average values during Phases 1 and 2 of IMMED.

In Classes A and B the nominal values used for BIOS-A were 0.0625 second during retrofire and 8.5 seconds thereafter; for Class C the nominal values were 0.125 second during retrofire and 20 seconds thereafter. For Class A trajectories the 0.0625-second retrofire integration step will be maintained, but a 30-second step down to 400,000 feet and 15-second-step thereafter will be used. Using these values, the impact point differs by 0.001 degree in latitude and by 0.003 degree in longitude from the nominal. Class B will use 0.125 second during retrofire and 30 seconds thereafter. This gives differences of 0.06 degree and 0.10 degree in latitude and longitude, respectively. Class C will use 0.5 second and 30 seconds, resulting in differences of 0.1 degree and 0.3 degree.

Class A and B changes have been successfully implemented and tested in NOMERL on the nominal orbit. The resulting error in impact was within the error described above. Further testing will be done on NOMERL on an eccentric orbit as well as in IMMED and FIXTIM. In IMMED, average values have been computed using the proposed Class C integration steps which compare with the nominal BIOS-A values as follows:

Station	Magnetometer Bias Constants (Milliamps)		OWSP Constants (Seconds)	
	Nominal	Proposed	Nominal	Proposed
FTMYRS	-1.196	-1.156	3039.9	3104.6
QUITOE	3.077	3.116	2953.6	2949.1
LIMAPU	4.175	4.175	3006.0	3006.0
SNTAGO	3.195	3.195	2872.3	2867.5
JOBURG	-1.117	-1.117	1256.9	1253.5

The greatest variations between the nominal and proposed magnetometer bias constants and OWSP constants are, respectively, one step and 65 seconds. Since the constants are average values these differences are not significant. Although further testing will be done on a total burn eccentric orbit, it appears the proposed integration steps for Class C may be used.

In conclusion, these changes should help to improve the response time for the call-down catalogs because much of the program running time is used in integrating through trajectories.

The greatest improvement in response time for the various call-down catalog routines will come by relaxing the criteria for selecting the time of retrofire. Mr. Robert Jackson of Ames Research Center has stated that hitting the point of closest approach to Hickam Field (165 degrees west meridian in the case of IMMED) is not as important as accurately determining the recovery point associated with a given retrofire. Thus, in NOMERL, FIXTIM, and IMMED (Phase 1), with an accurate guess of the initial retrofire, only one trajectory (which should land within one-half degree of the point of closest approach) need be run instead of the several formerly needed. In IMMED, Phases 2 and 3, this philosophy cannot be used because a yaw error angle exists at retrofire. Some preliminary thought has been devoted to finding a method for correcting the initial retrofire guess for yaw error angle, but thus far no satisfactory algorithm has been found. These changes have not been implemented in the call-down catalogs, but a test program has been written to check the subroutine which will compute accurately the initial retrofire. When this change is coupled with the increase in integration steps, the recovery programs could run up to 50 percent faster.

In the three call-down programs core has been reduced by removing the subroutines which compute the initial down-range table and the station acquisition from the call-down catalog programs. Both these routines are in STACON (which uses relatively little core). The initial down-range table and the station acquisitions are then input to the catalog programs via tape, thus saving core in those routines where it is needed. These changes have been coded in all the recovery programs and have been successfully tested in STACON and NOMERL.

System/360 Conversion

The recovery control programs have undergone some redesign in conversion to System/360. Since all these programs perform essentially the same initialization and read the same input parameter cards, an initialization routine has been written for this function. This routine will call any one or all of the main programs as determined by an input parameter. Thus, all the recovery control programs will be in core at the same time. This is possible due to the larger core memory of the System/360. The STACON option has worked, and the NOMERL option is being debugged. The subroutines which compute the impact point and the trajectory are being tested separately to facilitate the debugging process.

PROGRAM FOR NEXT REPORTING INTERVAL

The conversion of the present BIOS Recovery Control System programs to System/360 will be completed. Also, the final implementation and testing of the changes to increase program efficiency will be finished. Documentation of the 7094 version of the Recovery Control programs will be updated to reflect the present system and the System/360 version will be written. Finally, preparations will be made to support the BIOS-B launch and around-the-clock coverage will be provided for the three day mission.

Task 2

DOCUMENTATION

DISCUSSION

Thirteen documents were processed and delivered to NASA during the quarter. For these documents, technical writing support was provided in the form of editing, compiling, coordinating final typing, artwork, and assembling the final reproducible copy. Review copies of all documents were made available to the authors, and their comments and corrections were incorporated into the final manuscripts. Deliveries were made in the form of camera ready copy; however, proof copies were also delivered for each document.

Prelaunch analysis reports and launch day reports for the satellites TOS-B and OSO-E1 were completed and delivered to NASA. In addition, two volumes of program documentation, the "AE-B Attitude Determination System" and the "NIMBUS Ephemeris Programs," were delivered. Due to technical commitments by the authors, the program documentation volumes for the OSO-E1 and ATS-B satellites were not completed as anticipated.

The flowcharts of the entire ATS-B documentation volume were computer-generated by the Goddard Autodoc system. The system proved highly successful, providing approximately one week turn-around time between the time of rough draft submission and completion of final printout.

PROGRAM FOR NEXT REPORTING INTERVAL

A prelaunch analysis report and a launch day report for the satellite TOS-C will be prepared and delivered. In addition, three program documentation volumes (ATS-B, OSO-E1, and PERTAPE II) and the following publications are also scheduled for delivery:

1. Final Specification for a Generalized Data Analysis System
2. ATS-B Users Guide

Work will begin on the following:

1. Program documentation (System/360 version) of Biosatellite Recovery Control System
2. ATS-B Programming Specification for UNIVAC 1108
3. ATS-A Final Documentation

Further investigations of the computer-generated flowcharting techniques will be made, and the system will be used where feasible.

Task 3

ATS-B MANEUVER CONTROL

DISCUSSION

Launch Analysis Summary

An analysis of the successful December 1966 launch of spacecraft ATS-B (now ATS-1) was prepared and delivered to NASA (Ref 1). The analysis presents the results of apogee motor ignition, in-orbit maneuvers, and attitude determination for ATS-B, over the period 7 December 1966 to 15 January 1967. Decisions leading up to second apogee insertion into a near-synchronous drift orbit are reviewed as is the possibility of insertion at first apogee (this is of interest since first apogee ignition may be employed for later ATS spacecraft). Eleven in-orbit maneuvers utilizing the spacecraft hydrogen peroxide system (three for spin axis erection and eight for placing the spacecraft on-station) are described and a detailed description is given of ATS real-time attitude determination procedures.

Finally, the analysis details several conclusions and recommendations based on the experience gained during the December launch. The following paragraphs briefly amplify each of the above ATS-B launch analysis results (details are available in Ref 1).

Apogee Motor Ignition

The transfer orbit and spacecraft attitude achieved by ATS-B in its launch yielded close to nominal values, as shown by the data summarized in Table 2*.

*Table 2 orbital elements are the usual GSFC osculating set: semi-major axis (a), eccentricity (e), inclination angle (i), mean anomaly (M), argument of perigee (ω), ascending node right ascension (Ω); all quantities are dimensionless, with angles in radians and a in earth radii units (eru).

Table 2. Summary of ATS-B Transfer Orbit Estimates

Launch Date: 7 Dec. 66, Launch Time: $2^h12^m1^s$ GMT
Epoch (T_o): $2^h35^m21^s$ GMT (Agena-ATS Separation Time)

Orbit Estimates		GSFC Osculating Elements						Spin Axis Attitude		2nd Apogee Crossing Time (computed ignition time) GMT
Number	When available T_o+hrs	Semi-major Axis (eru) a	Eccentricity e	Inclination Angle (rad) i	Mean Anomaly (rad) M	Arg.of Per (rad) ω	Rt. Asc. of Asc. Node (rad) Ω	Rt. Ascen (deg) α_{SA}	Declin (deg) δ_{SA}	
Nominal	T_o+0	3.9026931	0.73637511	0.54240301	0.028138595	3.1359986	4 7529145	182.1	25.5	$18^h44^m18^s$
1	T_o+3 1/4	3.8943090	0.73730803	0.55178497	0.029675057	3.1070220	4 7676518	-	-	$18^h54^m15^s$ $(8^h10^m19^s)$*
2	T_o+4	3.3990326	0.73606220	0.54624885	0.029535212	3.1276387	4.7521086	-	-	$18^h46^m41^s$
3	T_o+11 3/4	3.8986923	0.73610184	0.5462453,	0.029520370	3.1274257	4.7521754	-	-	$18^h46^m39^s$
4	T_o+15 1/4	3.8986919	0.73610329	0.54628764	0.029520267	3.1274207	4.7521853	181.9	25.3	$18^h46^m39^s$ $(8^h01^m34^s)$*

*These represent, for comparison purposes, time of first apogee crossing.

This table compares the nominal transfer orbit elements against several sets of actual transfer orbit elements (determined by post-injection tracking measurements); nominal spin axis attitude, in terms of right ascension α_{SA} and declination δ_{SA}, and second apogee crossing times are also compared to corresponding actual values. (Nominal spin axis attitude is that required to give a post-apogee maneuver drift rate of 7 deg/day, while actual attitude is as inferred from an analysis of pre- and post-maneuver orbital data.) Consistent and near-nominal orbit estimates are seen to result after transfer orbit estimate 2 (only a two-second difference in computed ignition time—second apogee crossing time—is noted between orbits 2 and 4). To illustrate the degree of error in early orbit estimates—and the importance of delaying orbit maneuvers until reliable orbit estimates are available—Table 2 also gives the computed times of first apogee crossing for orbits 1 and 4. A nine-minute difference between these times is noted; such errors in ignition time can result in drift orbit inclination angle errors of about 0.6 degrees (Ref 2, pp. 61-64).

The data presented in Table 2 also shows that reliable orbit state estimates were available during the first ascent leg of the transfer orbit. Earliest transfer orbit estimates, available 3-1/4 hours after injection, were deemed unacceptable. However, orbit estimate 2, which was available about four hours after injection, differs only slightly from the firmer (and final) transfer orbit estimate 4, available 15-1/4 hours after injection. This means (since first apogee crossing nominally occurs about 5-1/2 hours after injection) that useful orbital state data was available before first apogee crossing.

The possibility of apogee motor ignition at first apogee crossing is of interest for future mission planning. Determination of the requirements for such a maneuver (i.e., ignition time and required spin axis attitude)—and the resulting GO/NO-GO decision for spacecraft reorientation prior to apogee motor ignition—demands accurate knowledge of both the orbital and attitude states of the spacecraft. Thus, ignition at first apogee depends upon the availability of firm orbital and attitude estimates.

As indicated above, acceptable orbital state data was available about 1-1/2 hours in advance of first apogee crossing. Although a two-hour period before ignition was originally estimated as necessary for preparation of station command data (in the event of a reorientation maneuver), experience acquired during the launch showed this estimate to be excessive; the 1-1/2 hour interval supplied by orbit estimate 2 was adequate. For the December 1966 launch, then, apogee motor ignition at first apogee was not limited by orbit determination considerations.

In Reference 1, it is noted that the quality of attitude data over the first three hours of the ATS-B transfer orbit was such as to return an acceptable attitude solution. However, two additional factors are present which influence the possibility of first apogee ignition, based on this attitude determination. The first relates to data handling and processing times. The attitude solution obtained using the first three hours of transfer orbit data was not actually in hand until about four hours after injection, because of time delays from actual measurement to transmission (20 minutes), from teletype reception through completion of editing (15 minutes), and from program run submission until returned solution (20 minutes). In real-time, then, the first transfer orbit attitude solution was not known until 0730, GMT, on 12/7/66—only about 1/2 hour prior to first apogee. This is considered an insufficient interval for preparation and execution of reorientation maneuver commands (had they been required).

The second factor pertains to station visibility and command capability. For the ATS-B transfer orbit, spacecraft elevation from ATS ground station Toowoomba at time of first apogee crossing was only about 4.3 degrees. The ability to achieve first apogee ignition is therefore influenced to whatever extent such low elevation angles affect station command capability. On the other hand, although ATS ground station Kashima had the spacecraft well in view at first apogee crossing (12 degrees elevation angle) and beyond, this station lacks command capability.

In summary, for the nominal ATS transfer orbit, ignition at first apogee is limited primarily by attitude determination and station command considerations, in particular:

- Delays associated with data handling after accumulation of sufficient measurements in real-time to permit confident attitude determination.
- Low elevation visibility of spacecraft from command station Toowoomba.

Spacecraft orbit determination is not a limiting factor. Finally, assuming command capability from Toowoomba is not lost at low elevation, first apogee ignition appears possible only in the event that a prior reorientation maneuver is not required.

As the spacecraft entered the ascending portion of the transfer orbit for the second time, current orbit estimates indicated that the next equatorial crossing (which corresponded essentially to second apogee passage) was to occ r to $18^h\ 46^m\ 39^s$, GMT. With this as an apogee motor ignition time, current attitude estimates indicated that a potential velocity savings of about 185 fps was possible if a reorientation maneuver was effected prior to ignition at second apogee. Such a maneuver normally requires a time consuming verification (and possible touch-up) procedure before commitment to fire. With second apogee fast approaching, this would have necessitated deferring ignition to a later apogee. Since ignition at third apogee involved an excessive increase in drift time to station, ignition delay past second apogee implied ignition at fourth apogee—and thus exposure of the apogee motor to its space environment for an additional 20+ hours. This increase in exposure time was felt to outweigh the potential velocity savings. The ATS Project Office thus decided to forego the reorientation maneuver and fire, with the current spacecraft attitude, at second apogee. Accordingly, the spacecraft was commanded to fire its apogee motor at $18^h\ 46^m\ 19^s$, GMT—20 seconds earlier than predicted equator crossing time (about one-half the nominal apogee motor burn time). This resulted in an actual ignition of $18^h\ 46^m\ 20.5^s$, GMT.

The near-equatorial orbit achieved by ignition at second apogee yielded results which proved to be closer to desired values than predicted, so that the decision not to perform a reorientation maneuver and to fire at second apogee proved sound. Predicted and achieved values of some important quantities associated with the apogee ignition maneuver are summarized in Table 3.

Table 3. Summary of Significant Apogee Ignition Maneuver Quantities

Quantity	Predicted Value	Achieved Value	Error (Achieved-Predicted)
Apogee Motor Velocity Inc, ΔV_B, fps	6145	6157	12
Apogee Motor Burn Time, T_b, sec	43	43.0	0
Spin Axis Rt. Ascension, α_{SA}, deg	181.4	181.9	0.5
Spin Axis Declination, δ_{SA}, deg	24.4	25.3	0.9
Resulting Drift-Rate, $\dot{L}$, deg/orb (+W)	10.7	7.5	-3.2
Resulting Inclination Angle, i, deg	0.75	0.23	-0.52

Note: Ignition time = $18^h 46^m 20.5^s$, GMT

Maneuvers

Spin axis erection and transfer of the spacecraft to its on-station location required eleven maneuvers. Three of the maneuvers were required for attitude control and eight for orbit control. Each of these maneuvers is described in detail in Reference 1. Table 4 shows the initial spacecraft characteristics which were used for the calculations detailed in the reference; Table 5 gives a condensed summary of all maneuvers.

Table 4. Initial Spacecraft Characteristics used for Maneuver Calculations

Initial Weight (after apogee motor fire)	774.27 lbs
Initial Peroxide System Weights	
System A	44 lbs
System B	82.73 lbs
Moment of Inertia-I_{roll}	
Initial	82.25 slug ft^2
Final (with all H_2O_2 expended)	67.54 slug ft^2

The orbit control maneuvers proved successful in that the spacecraft arrived at its designated on-station location with the desired drift rate and a relatively low eccentricity. However, eight maneuvers were required to achieve these ends. The main reason for this large number was the uncertainty which existed with the peroxide systems after the B system axial jet malfunctioned (see Ref 1). The second was the difficulty of getting agreement between predicted and actual orbits when small corrections were attempted.

The exact cause of the difficulty is not known at this time. There is a natural tendency to blame the peroxide system for the problem since the number of pulses used for some of these maneuvers was relatively small (a small

Table 5. Summary of Maneuver Calculations

Maneuver Number	Type	Date	Time (GMT)	System	Est. Vel. Change	Result Drift (deg/day)	Resulting Attitude** Rt. Asc. (deg)	Resulting Attitude** Dec. (deg)	Fuel Used (lbs)	Pulses
1	REOR	12-08-66	18:30:00	B	—	—	—	—	5.73	2911
			20:10:30	A	—	10.676W	3	-77	1.53	669
2	REOR	12-09-66	16:30:00	A	—	10.658W	11	-88	0.82	301
3	ΔV	12-12-66	22:35:00	B	7.5	9.875W	11	-88	1.05	498
4	ΔV	12-13-66	02:23:00	B	—	—	11	-88	3.82	1360
5	ΔV	12-13-66	03:30:00	A	—	5.587W	11	-88	3.09	1360
6	ΔV	12-14-66	18:26:40	B	9.2	4.658W	11	-88	1.8	638
7	ΔV	12-15-66	07:29:00	B	18.2	2.666W	11	-88	3.33	1377
8	ΔV	12-16-66	08:10:27	B	16.7	0.825W	11	-88	3.09	1298
9	ΔV	12-16-66	22:07:00	B	12.5	0.1017W	11	-88	1.89	976
10**	ΔV	12-20-66	00:20:00	B	1.0	0.0159E	10	-87.5	0.17	83
11	REOR	12-22-66	20:00:00	A	—	0.0147E	8	-89.3	0.1	44

— Not determined

* Initial Drift: 7.496W

Initial Attitude: $\alpha = 180.^{0}6$; $\delta = +23.^{\circ}3$

** Values are averages taken from several attitude determinations.

number of pulses is supposed to increase peroxide engine inaccuracies). Also, the accuracy of the direction of the impulses is open to question since no simple verification method is available. However, magnitude determinations based upon orbital period measurements were within eight percent of predicted, and the attitude maneuvers indicate that no large direction errors were present.

Another possibility for the source of trouble is the orbit itself. Firm assurances of accurate elements are required for cases of small eccentricity and inclination. This is important because it was always assumed that maneuvers were being performed at an apse, yet in every case there was some indication of a radial component which shifted the perigee and/or altered the eccentricity less than desired.

A third possibility is the manner in which the maneuvers are planned. The program which computes the direction of the pulses (i.e., the jet-start-angle) is based upon an impulsive thrust approximation. In practice, this may not be the situation since some of the corrections required up to 15 minutes for completion, during which time the velocity vector rotated approximately 4 degrees. The above, coupled with any comparable errors in the peroxide system, could explain the difficulties experienced in correlating measured and predicted results.

Spacecraft attitude was changed three times by means of Maneuvers 1, 2 and 11 and resulted in a final spin axis declination of -89.3 degrees. While this may seem acceptable, an examination of Maneuvers 2 and 11 shows relative errors of between 20 and 30 percent with respect to the desired magnitude of change. Maneuver 1 is not considered in this evaluation because of the uncertainty caused by movement in the wrong direction and malfunctioning of the B system axial jet (it is a credit to the spacecraft that the spin axis arrived as close to the south celestial pole as it did after the occurrence of these two events).

Maneuver 2 was performed with an uncalibrated peroxide system; this partly explains the large relative error in the result. For Maneuver 2 this error was based upon a desired motion of 12 degrees against a miss distance of 2.5 degrees. In addition, Maneuver 2 had to contend with noisy POLANG data. Such data made it difficult to determine reasonable attitude for the touch-up maneuver.

Maneuver 11 was one of small magnitude (2.5 degrees) but the miss distance was 0.7 degrees. Since no touch-up was performed, Mercator chart or POLANG errors did not contribute to the miss. However, for maneuvers of this size it is felt that the uncertainty of the initial position (±1 degree) plus incorrect calibration was the primary cause of the large relative error.

As a result of these eleven maneuvers, some indication was obtained of the performance of t. our peroxide engines. In-orbit calibration of the radial jets was based upon determining the ratio between actual and predicted orbital period changes. The period was chosen for calibration purposes for two reasons: first, orbital period can be accurately measured and, second, small period changes are linear functions of small changes in velocity vector magnitudes. No other calibrations were made (e.g., velocity increment direction was not checked since no detection means were available at the time of the maneuver; the only applicable computer program gave unreliable results for velocity increments less than 50 fps).

The performance of the operational axial jets is based upon graphical measurements obtained from Maneuver 2 and is assumed to carry over to the second time it was used (Maneuver 11). A summary of the performance evaluation of each engine is given in Table 6.

Table 6. Hydrogen Peroxide Engine Performance

System A	Engine Operating Limits (percent of nominal)
Axial	± 10
Radial	not measured
System B	
Axial	-83
Radial	±8

Attitude Determination

Reference 1 summarizes ATS-B attitude history during the transfer orbit and after erection to the orbit normal, as determined from various mixes of station-acquired data by the attitude determination program ATTDET. Overall quality of data by type and station, as received during the mission, is discussed in terms of both short-term smoothness and long-term variation when compared with predicted behavior. A series of postflight program test runs which combine the data in different ways for purposes of error evaluation is also described. These tests are mainly confined to the transfer orbit phase, and assume as a standard of comparison or "true" attitude an independent solution based on pre- and post-apogee fire orbits. Although results to date are not sufficient to establish exact error levels for the data from each station, some approximate error bounds were determined and are reported in the cited reference.

Table 7 summarizes the attitude history determined by ATTDET (in terms of spin axis right ascension, α, and declination, δ) during the ATS-B transfer and drift orbit phases, up to the time of "final" attitude erection to orbit normal. The data mix by type and station is included for each run with the approximate GMT span of the POLANG* data. Stations are identified by number (in parentheses) according to the following key:

1. Toowoomba　　2. Rosman
2. Mojave　　4. Kashima

Sun data is specified as to origin by ACO (onboard Angle Counter Output) or SC (ground synchronous controller).

Some comments on the runs reported in Table 7 include the following:

- The several bad Toowoomba POLANG points contained in Runs 1 and 2 were graphically detected. It was subsequently determined that removal of these points in Run 1 would have resulted in a solution of about $\alpha = 182.9$, $\delta = 27.0$ degrees.

*POLANG (for polarization angle) is defined as the dihedral angle between the plane containing the station-to-spacecraft line-of-sight and station zenith vectors and the plane containing the line-of-sight and spin axis vectors (see Figure L-2 of Reference 1).

Table 7. Attitude History from Transfer Orbit to Final Erection

Run #	α(deg)	δ (deg)	Sun Data	POLANG Data	Remarks
1	184.0	28.7	(1) ACO	(1) 12/7/66: 0335-0407	Transfer orbit-early Toowoomba data only.
2	183.1	25.3	(1) SC	(1) 12/7: 0335-0620 (4) 12/7: 0457-0608	Transfer orbit-Toowoomba and Kashima data.
3	182.7	24.6	—	—	Similar to #2 but with several bad Toowoomba POLANG points removed.
4	181.4	24.4	(1) ACO	(1) 12/7: 0620-0741 (2) 12/7: 1429-1449 (4) 12/7: 0457-0837	Transfer orbit-Toowoomba, Rosman, and Kashima data. This solution adopted for second apogee motor fire computations.
5	181.4	24.8	—	—	Similar to #4 but with Kashima data deleted.
6	180.6	23.3	(2) ACO (3) ACO	(2) 12/7: 2030-12/8: 1151 (3) 12/7: 2132-2333	Drifting orbit after apogee motor fire and prior to erection maneuver. First erection maneuver based on this solution.
7	3.2	-76.9	(2) ACO (3) ACO	(2) 12/8: 2021-12/9: 0002 (3) 12/8: 2021-12/9: 0915	Drifting orbit after first erection maneuver. Second erection maneuver based on this solution.
8	11.2	-87.8	(2) ACO (3) ACO, SC	(2) 12/10: 1630-12/12: 1430 (3) 12/11: 2201-12/12: 1100	Drifting orbit after second erection-distilled POLANG data-Rosman data manual.
9	10.7	-88.4	(1) ACO (3) ACO	(1),(2),(3) 12/12-12/14	Drifting orbit after 12/12 slow down-distilled POLANG data.
10	9.3	-87.3	(1) SC (3) ACO	(1) 12/20: 0030-1530 (2) 12/21: 1735-12/22: 1630 (3) 12/20: 0810-12/22: 1203 (4) 12/20: 0130-12/22: 0435	Drifting orbit-distilled POLANG from all four stations-last run before final erection 12/22.

- In Solution 4, initial Rosman data received on approach to second apogee was considered to reasonably confirm Run 3 estimates based on Toowoomba and Kashima data. This solution was therefore adopted as input for second apogee motor fire calculations.

- The purpose of Run 5 was to investigate possible existence of Kashima POLANG bias, by comparison of results with Run 4. Since only a minor difference of 0.4 degrees in the δ solution was observed, the test was considered negative.

- The attitude change in Run 6 is a probable result of perturbations encountered during apogee motor fire.

- The distilled data referred to in Rune 8-10 was obtained by replacing every POLANG message of ten values over a ten-second interval by a single averaged value at the midpoint time.

- For the final erection maneuver, a composite attitude estimate of $\alpha = 10.0$, $\delta = 87.5$ degrees as deduced from Runs 8-10 was employed.

Table 8 summarizes ATTDET attitude solutions obtained during the post-erection on-station mission phase (through 11 January 1967). In general, POLANG data from all stations following final erection on 22 December 1966 has tended to be erratic compared with data received prior to that time. The solutions reported in Table 8 are therefore based quite heavily on sun data (and in particular on its time variation over each solution interval). Only ACO type sun data were available for these runs.

Table 8. Posterection Attitude Runs

Run#	α(deg)	δ(deg)	ACO Data	POLANG Data	Data Interval
11	14.3	-88.9	MOJ	ROS, MOJ, KAS	12/23 - 12/26
12	6.4	-88.2	ROS	ROO, ROS, MOJ	12/27 - 12/30
13	0.0	-89.3	MOJ	ROS, MOJ	1/4 - 1/5
14	18.0	-89.2	MOJ	--	1/3 - 1/8
15	5.1	-89.0	MOJ	ROS, MOJ	1/3 - 1/8
16	8.0	-89.3	MOJ	ROS, MOJ, KAS	1/3 - 1/10
17	7.5	-89.3	MOJ	TOO, ROS, MOJ, KAS	1/3 - 1/11

The fluctuations in α noted in the table result from high sensitivity of α to data errors near the 90-degree declination point (where α becomes indeterminate). From the above runs, a combined solution of $\alpha = 8.0$, $\delta = -89.3$ degrees was adopted for the posterection attitude as of 11 January 1967.

The quality and consistence of POLANG data has shown considerable variation during various phases of the mission. As a general statement applicable to all stations, the smoothest POLANG data was received during the transfer orbit. Error fluctuations of varying magnitudes then appeared during the drift orbit up to the time of final erection. Following on-station erection, POLANG has been received sporadically, and is often so uncertain as to be unuseable.

Sun data of both ACO and SC types have been regularly obtained throughout the mission, with ACO points usually dominating in number. Although these measurements are in theory identical, ACO data are received only to the nearest 0.1 degree, so that accurate values are available only when the angle is noted to change incrementally by 0.1 degree. Yet even at such reference points, the SC values (which as recorded are supposed to be accurate to 0.01 degree) have consistently differed from the ACO readings. This difference has been in the range 0.2 to 0.5 degree during the entire mission. Although some tests indicate the ACO data is the more reliable, other tests indicate only small differences in attitude solutions employing ACO or SC data separately.

Since arrival of the spacecraft on-station, a number of postflight evaluations have been performed in an attempt to establish firmer quantitative levels for attitude data accuracy during the mission. The major part of this work has been directed toward detecting possible POLANG biases for each station. Such biases can exist because of calibration and/or Faraday rotation errors, and for reasonably short data intervals can be considered approximately constant.

To implement the above, a test sequence (detailed in Ref 1) was carried out using distilled and edited data acquired during the ATS-B transfer orbit. Attention was confined to this phase of the mission because an independent attitude solution was available and because of the comparative smoothness of POLANG data for this period (this constitutes a favorable condition for possible bias detection). Results of these tests are given in Table 9.

Table 9. Bias Detection During Transfer Orbit

Run	Data	α(deg)	δ(deg)	POLANG Bias (deg)
1	ACO + Rosman and Toowoomba POLANG	181.7	24.8	Not estimated
2	ACO + Rosman and Toowoomba POLANG	181.5	24.5	-0.7 (Toowoomba)
3	ACO = Rosman and Mojave POLANG	181.6	24.5	Not estimated
4	ACO + Rosman and Kashima	181.7	24.5	-0.1 (Mojave)
5	ACO + Rosman and Kashima POLANG	181.3	24.3	Not estimated
6	ACO + Rosman and Kashima POLANG	181.4	24.5	+0.5 (Kashima)

Assuming Rosman to be unbiased during this phase, a creditable bias estimate for another station is one which, when included in the solution, results in spin axis α, δ again equal to the Rosman-alone attitude. In this sense the Toowoomba and Kashima bias estimates of Table 9 are seen to be consistent, since the α, δ solutions with bias estimates return close to the Rosman-only values of 181.4, 24.5. Moreover, the solutions of runs 2 and 6 of Table 9 have been closely duplicated (within 0.1 degree) by Toowoomba-only and Kashima-only runs to which fixed POLANG bias values of -0.7 and +0.5 degree, respectively, were added. Nevertheless, the bias estimates obtained are not at this time considered entirely conclusive, since with bias not estimated (Runs 1 and 5), the attitude solutions were already in fair agreement with the Rosman reference. The bias test for Mojave is also not conclusive; but if present, the bias is probably insignificant.

The above type of test was pursued further by examining a situation which occurred around December 20 through December 22 during the drift orbit. From the study of certain auxiliary runs (not containing Kashima data) made just prior to Run 10 of Table 7 (which includes Kashima data), and comparison of these runs with Run 10, some Kashima bias was suspected. Assuming the other three stations to be unbiased at this time, the second and third runs of Table 10 were made in an attempt to isolate any Kashima bias.

Table 10. Kashima Bias Detection During Drifting Orbit

Data	α(deg)	δ(deg)	POLANG bias (deg)
Run 10, Table 7 (all stations)	9.3	-87.3	Not estimated
Run 10, with Kashima data deleted	2.7	-87.0	Not estimated
Run 10, with Kashima data deleted	2.6	-87.0	2.3 (Kashima)

It is seen here that a nearly identical attitude solution results from either deletion of Kashima data, or inclusion of these data together with Kashima bias estimate. Thus, the evidence is strong that a Kashima bias of around 2.3 degrees existed during the drift orbit, if indeed the other three stations were then unbiased. (Further testing is required to substantiate this assumption.) Also, a change in Kashima bias from what may have existed during the transfer orbit (Table 9) is not per se contradictory, since both calibration and Faraday rotation correction errors may change with time.

With regard to sun data, two types of tests have been conducted to assess accuracy levels. First, an after the fact attempt was made to determine the transfer orbit attitude from sun data alone. Results from ATTDET were negative using either ACO or SC data alone or in combination—the program simply failed to converge. This is attributed to the relatively short time span (about 16 hours) of the data, over which sun angle geometry undergoes only a very small change. Second, various runs with combined POLANG and sun data have been made to determine which type of sun data is more accurate, since experience to date indicates a typical 0.4 degree bias-like difference between the two values. As mentioned above, it was found that the transfer orbit attitude solution was insignificantly affected by the use of SC rather than ACO data. Another potentially useful period for comparison is during the on-station orbit. ACO data has been regularly received over this phase, and because of the long time spans involved (days rather than hours), it has in fact been possible to obtain attitude solutions in close agreement with Runs 14-17 of Table 8 by the use of

ACO data alone. Unfortunately, SC data has been received only irregularly during the on-station orbit, thus precluding additional comparative solutions.

ATS Attitude From Soumi Camera Pictures

The Soumi camera pictures can be used to determine the ATS-B spin axis attitude. This was demonstrated to reasonable accuracy by D. Fordyce of the NASA ATS Project Office. This effort has been continued and at present the technique is being automated for increased accuracy. A qualitative description of the method, including an error analysis is given in the following paragraphs; exact details will appear in the program documentation.

Soumi Camera Picture

The Soumi camera aboard ATS-1 has rtical field of view of approximately 7.6 degrees. This vertical field is scanned .017 lines with the horizontal sweep provided by spacecraft spin. Since the vertical field is less than the angle subtended by the earth at synchronous altitude, chords appear on the resulting photograph. If the camera center line exactly pierces the center of the earth, the upper and lower chords are of equal length; if the camera center line does not pierce the earth center, the two chords are of unequal length. Also, if this center line moves periodically with respect to the earth center this motion is reflected in the earth center to chord line distance. Since the spin axis is fixed in inertial space, any periodic displacement is caused by the spacecraft orbital motion. Figure 1 shows these effects.

To determine attitude, spin axis tilt from the orbit normal is first found. This tilt corresponds to the maximum magnitude of the (periodic) function which describes the change in the distance from earth center to chord line. Spin axis position in inertial space is then obtained by coordinate rotation from an orbital plane frame to the inertial frame. The point in the orbit where this is performed is governed by the time when the distance from earth center to chord is maximum. These points are expanded upon in what follows.

The first item investigated was whether the picture as determined from the camera is a portion of a circle. To do this it is only necessary to show that the distance from picture center to earth limit is constant. Figure 2 shows the

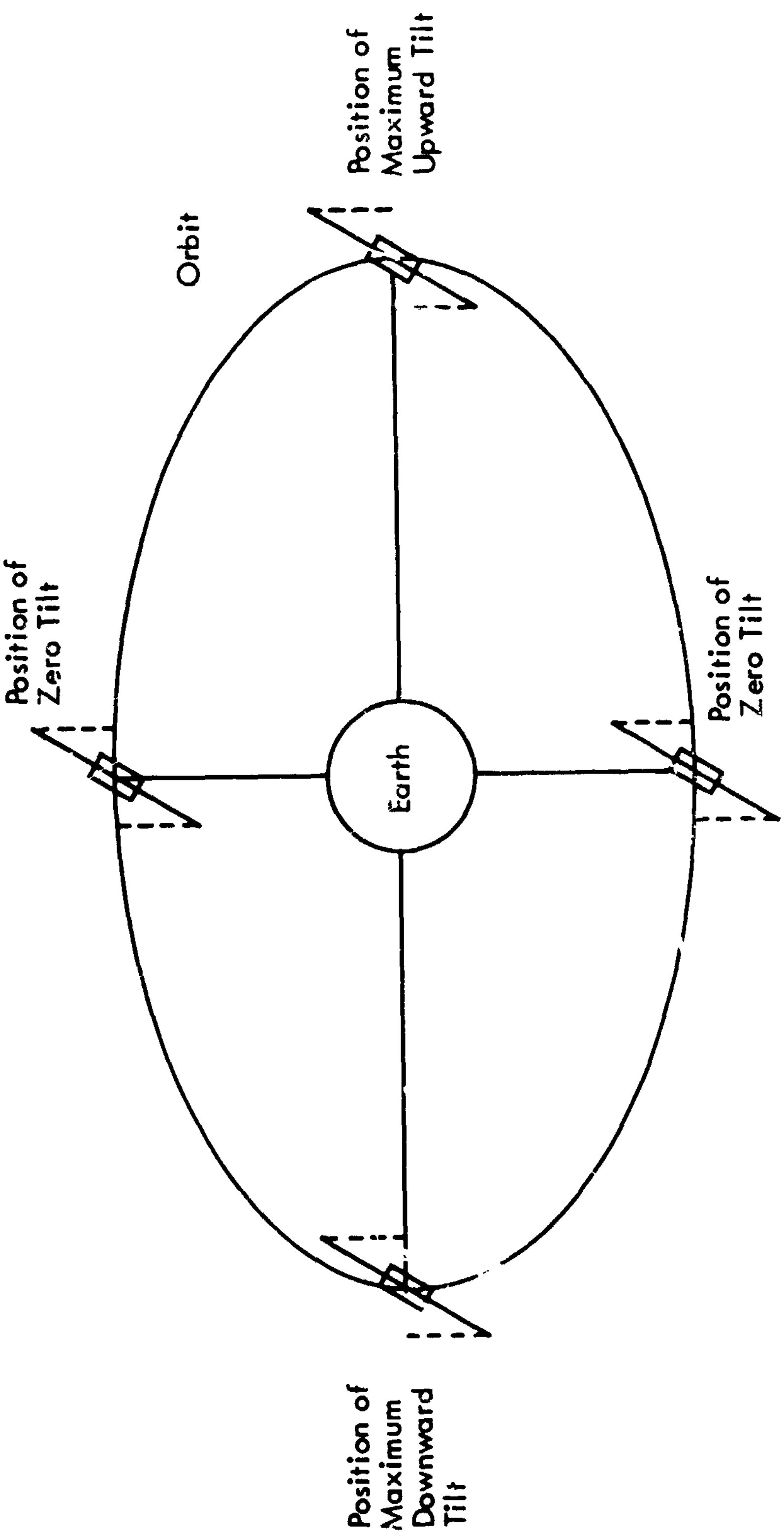


Figure 1. ATS-B Camera Tilt with Respect to Orbit Position

geometry involved. The angle α corresponds to the vertical scan angle; this angle varies linearly in 2017 steps from the top to the bottom chord. The angle β corresponds to the horizontal sweep; since the spacecraft spin rate is constant, this angle varies linearly from the left to the right limb. From this it can be obviously concluded that the picture has rectangular coordinates in angle measurement (degrees or radians).

The point where the line of sight is tangent to the earth (assumed spherical) is shown at point T. In the three dimensional geometry of Figure 2, spacecraft-centered spherical coordinates are designated by (a, β, α), where a is the distance from spacecraft to tangent point and α and β are as described above. Note that since each horizontal line is obtained at a constant α angle, points on the limb have greater latitudes than those at the center. This is because the camera scans a three-dimensional figure and not its two-dimensional projection.

At the point of tangency, T, a right triangle is formed. This triangle will be the same for every point of tangency (assuming a spherical earth) so that the direction cosine of T, with respect to the line to the earth center (X-axis), will be a constant angle Θ where

$$\Theta = \sin^{-1}\left(\frac{R_E}{r}\right)$$

Using the above and the direction cosine definition one obtains

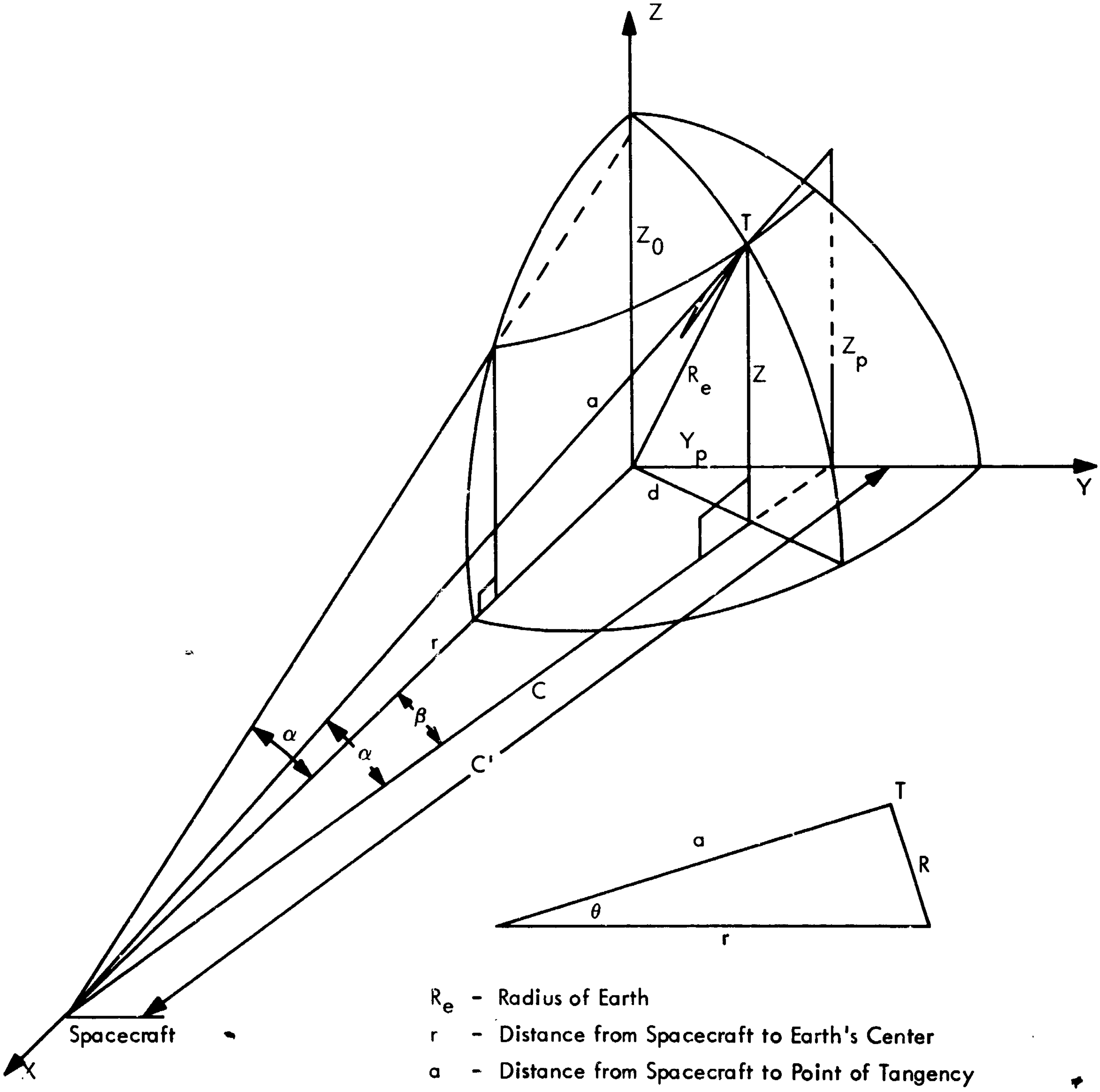
$$\cos \alpha \cos \beta = \cos \Theta = \text{constant}$$

Now since α and β are relatively small the left side can be expanded as

$$\left(1 - \frac{\alpha^2}{2}\right) \quad \left(1 - \frac{\beta^2}{2}\right) = \cos \Theta$$

or

$$\alpha^2 + \beta^2 = 2\ (1-\cos \Theta) + \frac{\alpha^2\beta^2}{2}$$



R_e - Radius of Earth
r - Distance from Spacecraft to Earth's Center
a - Distance from Spacecraft to Point of Tangency
T - Point of Tangency
α - Vertical Scan Angle of Camera
β - Horizontal Sweep Angle of Camera
Z - Axis Parallel to Spin Axis

Figure 2. Spacecraft Camera Viewing Geometry

The above shows that if $\frac{\alpha^2\beta^2}{2}$ is neglected, the earth image should indeed photograph as a circle. To demonstrate that $\frac{\alpha^2\beta^2}{2}$ can be neglected, representative values can be substituted into the above. Since $\frac{\alpha^2\beta^2}{2}$ is a maximum when $\alpha = \beta$, this point will be investigated. First assume that $\Theta = 8.7$ degrees and $\cos\alpha \cos\beta = \cos^2\alpha = 0.98849$, giving $\alpha = 0.10734$ and $\frac{\alpha^2\beta^2}{2} = 0.00006688$. If the radius is considered as the square root of the right hand side of the equation then

$$R = \left[2\,(1 - \cos\Theta) + \frac{\alpha^2\beta^2}{2}\right]^{1/2}$$

Substituting values gives

$$\begin{aligned} R &= (0.0230 + 0.00006688)^{1/2} \\ &= 0.0230\,\left(1 + \frac{0.00006688}{.0230}\right)^{1/2} \\ &= 0.0230\,(1 + 0.00145), \text{ radians} \end{aligned}$$

The above shows that the picture deviates from a circle when $\alpha = \beta$ by approximately one part in seven hundred, or less than 0.2 percent. If the pictures have a radius of four inches and can be measured to 0.01 inch (as an outside limit) the deviation will then be 0.006 inch at $\alpha = \beta$; this is well below the measurement accuracy and should not be detected.

A possible area for further study is the effect of the oblate earth on the above. However, the earth's ellipticity is 0.0034 which is the same order of magnitude as the above; the error will thus be approximately the same and will occur at the pole and not at $\alpha = \beta$. The conclusion, therefore, is that the circular picture assumption is well justified.

Distortion

Using the assumption that the camera picture should appear as a circle, any observed distortion can then be attributed to picture processing. In the following paragraphs it will be assumed that when picture data are displayed on a CRT the horizontal and vertical sweeps each have slightly different gains. If these amplifiers are linear the circle should then be distorted into an ellipse and it is this fact which will be used in processing the measurements taken from the picture.

The measurement desired from each picture is the distance from the center to the lower chord. This chord is in the southern hemisphere and is henceforth called the Southern Frame Limit (SFL). This distance must then be related to some angular measure or scale factor. The angular measure in each case will be the half-angle subtended by the earth's equator at the spacecraft. Because of the eccentricity of the orbit this will vary for each picture. However, since the orbit is well known, predicted values of this distance are easily obtainable and will be included in all calculations. The scale factor is therefore (see Figure 2)

$$\Theta = \sin^{-1}\left(\frac{R_e}{r}\right), \text{ degrees}$$

where

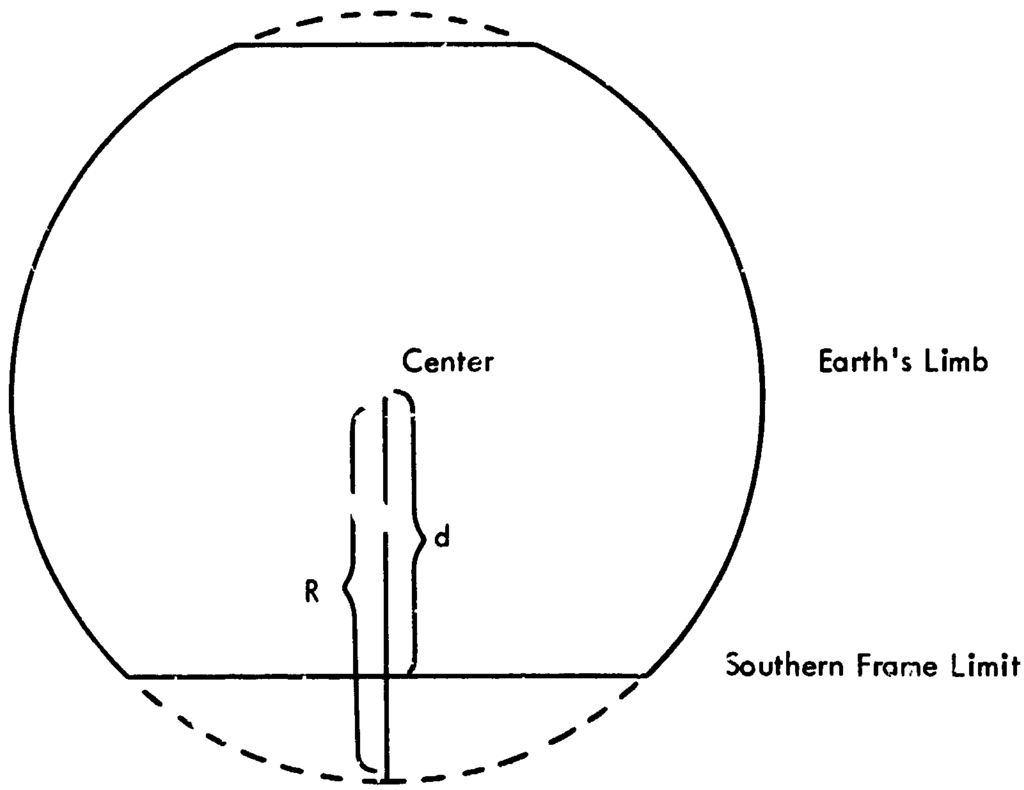
R_e = earth equatorial radius

r = distance from earth's center to spacecraft

To obtain the angular distance from earth center to SFL, the ratio of this distance to the associated major or minor axis of the ellipse is first determined (see Figure 3). This ratio is then multiplied by the above scale factor to obtain the desired quantity. This procedure eliminates any linear distortion and also includes the effects of an eccentric orbit in the scale factor.

Picture Shape Determination

To determine picture shape characteristics, each picture is scanned by a measuring device (the Concord S-5 LINE machine) which determines the Cartesian coordinates of the visible limb of the earth. By noting the coordinates of the limb—



d - Distance from Center to Southern Frame Limit

R - Semi-Axis of Conic in Direction of d

Dotted: Not seen in Picture

Figure 3. Measurements to be Made on Soumi Camera Picture

SFL intersection, the data are then processed in a least squares sense to determine the coefficients of the general conic equation:

$$Ax^2 + By^2 + Cxy + Dx + Ey + 1 = 0$$

The condition equations are linear in the constants A, B, C, D and E. After these constants are determined, the necessary translation and rotation can then be effected to cast the ellipse equation in standard form

$$\frac{x^2}{a^2} + \frac{y^2}{b^2} = 1.$$

The SFL distance is then associated with one of the above ellipse axes, depending upon how the coordinate systems are defined (see Figure 3).

Attitude Dete. mination

The relationship between spin axis, orbit normal, and orbit position are shown in Figure 4. The angle η is the angle between the spin axis and position vectors. In the orbit plane coordinates shown the scalar product is

$$\cos \eta = \cos (90 - A) \cos (v - \phi_1)$$

where

A = spin axis tilt from orbit normal

v = true anomaly measured from a convenient reference (not perigee)

ϕ_1 = arbitrary phase angle

If η is changed to $(90 - \epsilon)$ and the left side expanded the tilt becomes

$$\epsilon = A \cos (v - \phi_1)$$

Since the pictures supply the distance from earth ce..er to SFL, a constant must be added to each side of the above to give

$$\epsilon + D = D + A \sin (v-\phi).$$

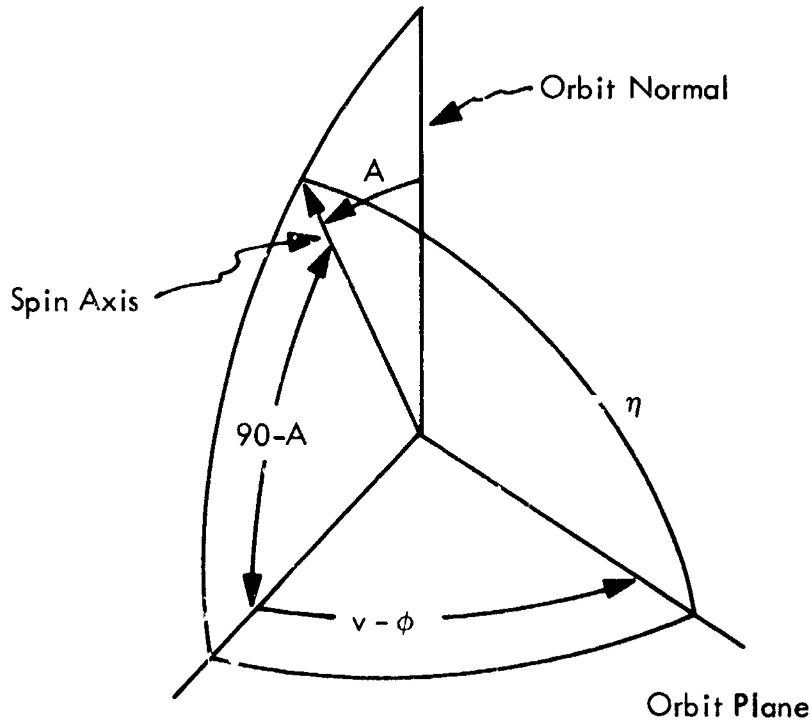


Figure 4. Relationship between Orbit Normal, Spin Axis and Orbit Position

The phase angle ϕ now includes the 90 degrees required for a computationally-convenient cosine to sine change. The quantity on the left side of the above equation is measured as (see Figure 3)

$$y \equiv \epsilon + D = \frac{d}{R}\theta$$

If the right side of the same equation is expanded

$$\frac{d}{R}\theta = D + B \sin v + C \cos v$$

where

$$B = A\cos\phi$$
$$C = -A \sin\phi$$

The measurements can then be fitted to the above by least squares and the quantities A and ϕ are thus determined. The value of A gives spin axis tilt from the orbit normal direction.

To obtain the inertial attitude, a rotation must be performed from the orbit plant frame to the inertial frame. When $v-\phi$ equals 90 degrees, the spin axis projection on the orbit plane is coincident with the position vector; it is also directed positively with respect to the position vector. Figure 5 shows this configuration. If a coordinate system is defined such that X is along the position vector, Z along the orbit normal, and Y orthogonal to X and Y (right-handed) the coordinates of the spin axis in this system are then

$$\vec{S} = \begin{bmatrix} \sin A \\ 0 \\ -\cos A \end{bmatrix}$$

As the orbit is known the value of the true anomaly can be found. The necessary rotations for placing the spin axis in the inertial frame are thus defined; from this obviously follow the desired spin axis right ascention and declination.

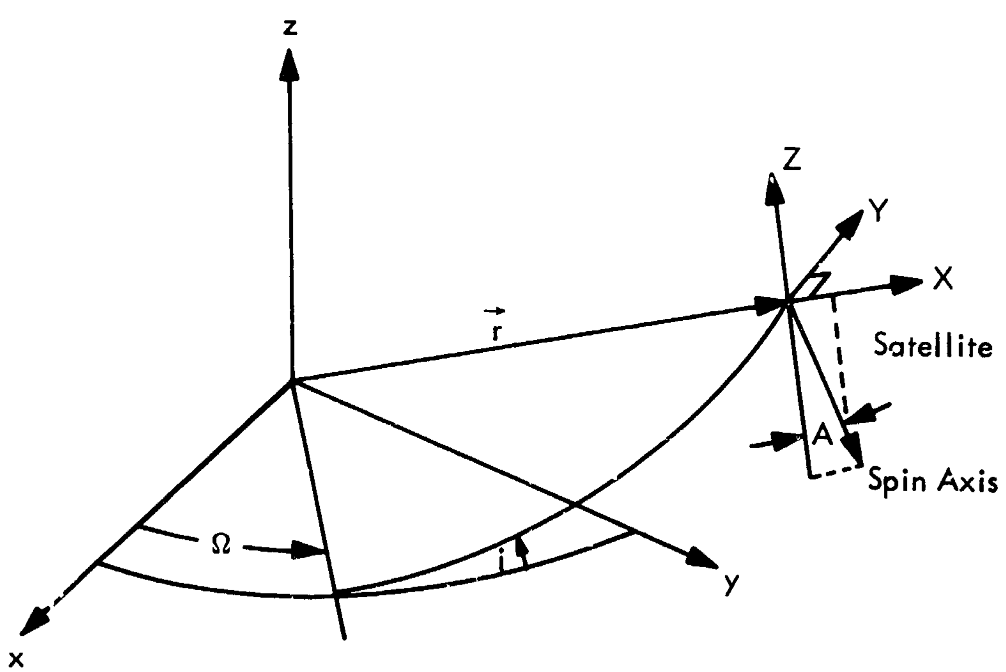


Figure 5. Relationship between Orbit Plane and Inertial Coordinates

Error Analysis

Spin axis tilt from the orbit normal is essentially determined by measuring two distances on each photograph and multiplying the ratio of these distances by an appropriate scale factor. This procedure forms the basis for an error analysis and, expressed as an equation, becomes

$$y = \frac{d}{R}\Theta$$

where

d = distance from picture to SFL

R = distance from picture center to earth limb in direction of d

Θ = scale factor in radians $= \sin^{-1}\left(\frac{R_e}{r}\right) \approx \frac{R_e}{r}$

R_e = earth's equatorial radius

r = distance from spacecraft to earth center

It is assumed in what follows that all individual error sources are independent so that

$$\sigma_y^2 = \left[\frac{1}{R}\cdot\frac{R_e}{r}\right]^2 \sigma_d^2 + \left[\frac{d}{R_e^2}\cdot\frac{R_e}{r}\right]^2 \sigma_R^2 + \left[\frac{d}{R}\cdot\frac{1}{r}\right]^2 \sigma_{R_e}^2$$

$$+ \left[\frac{d}{R}\cdot\frac{R_e}{r^2}\right]^2 \sigma_r^2$$

To form an estimate of the error in y, realistic assumptions will be made for the various quantities in the above brackets and for the associated variances. These assumptions are, for the bracketed quantities,

$$\frac{R_e}{r} = \frac{3441 \text{ nm}}{22752 \text{ nm}} = 0.15124$$

$$\frac{d}{R} = 1$$

$$\frac{1}{R} = \frac{1}{400} = 0.0025$$

Some comments are in order for the last two of these. Since the greatest error occurs when the SF L approaches the limb, a worst cast situation is given when this ratio is unity. It is also assumed that distances can be measured to 0.01 inch and, since the photographs have an earth radius of approximately 4 inches, the magnitude of R is 400. Using these values, the above equation becomes

$$\sigma_y^2 = \left[14.296 \times 10^{-8}\right] \sigma_d^2 + \left[14.296 \times 10^{-8}\right] \sigma_R^2$$
$$+ \left[0.19318 \times 10^{-8}\right] \sigma_{R_e}^2 + \left[0.0044187 \times 10^{-8}\right] \sigma_r^2$$

If the errors in d and R are equal (a valid assumption) the equation becomes

$$\sigma_y^2 = \left\{28.592\, \sigma_d^2 + 0.19318\, \sigma_{R_e}^2 + 0.0044187\, \sigma_r^2\right\} 10^{-8}$$

The variances are now assumed to be given by

$$\sigma_d^2 = (1)^2 = 1$$
$$\sigma_{R_e}^2 = (5)^2 = 25\ \overline{\text{nm}}^2$$
$$\sigma_r^2 = (1)^2 = 1\ \overline{\text{nm}}^2$$

The variance in d is based upon the measurement resolution of the device which scans the picture (0.01 inch). The variance of R_e, the radius of the earth, is relatively large and chosen as such because of cloud height uncertainty. The variance in r, the spacecraft distance, is large but since it has such a small effect it is so chosen for computational convenience. The total variance is then, finally,

$$\sigma_y^2 = \{\ 28.592 + 4.8295 + 0.0044187\ \}\ 10^{-8}$$
$$= 33.4259 \times 10^{-8}$$

and the standard deviation is

$$\sigma_y = 5.7815 \times 10^{-4} \text{ radians}$$
$$= 0.03313 \text{ degrees}$$

The use of least squares to determine a final tilt value tends to further reduce the above by a factor of three. This gives a final estimated error or approximately 0.01 degree.

To determine how the above error reflects in spin axis declination, the coordinate transformation effects must be examined (see Figure 5). Since the spin axis z inertial coordinate is related to the declination by

$$\sin \delta = z$$

this relation is used for the analysis. Thus,

$$\cos \delta \Delta\delta = \Delta z$$

or

$$\left(\frac{\pi}{2} - \delta\right) \Delta\delta = \Delta z$$

The inertial z coordinate is obtained (Figure 5) as

$$z = \sin A \sin \omega \sin i - \cos A \cos i$$

$$= A \sin \omega \sin i - \cos A \cos i$$

Taking the differential and assuming $\omega = 90$ degrees

$$\Delta z = \sin i \Delta A + \sin A \cos i \Delta A$$

Also, since i is close to zero

$$\Delta z = (i + A) \Delta A$$

In the above, the assumption is made that the error in inclination angle i is negligible compared to that in tilt angle A. The total variance for the declination is then

$$\sigma_\delta^2 = \left[\frac{i + A}{\frac{\pi}{2} - \delta}\right]^2 \sigma_A^2$$

Note that in the situation used above, $\omega = 90$ degrees so that

$$i + A = \frac{\pi}{2} - \delta$$

and

$$\sigma_\delta^2 = \sigma_A^2$$

The same will apply for $\omega = 0$, so that the tilt error appears in the declination on a one-to-one basis.

The error in right ascension is also affected by inaccuracies in the time of the maximum SFL distance. This caused by the error in the phase angle ϕ. To examine the right ascension error the following expression is used

$$\tan \alpha = \frac{y}{x}$$

so that

$$\Delta\alpha = \frac{x\Delta y - y\Delta x}{x^2} \cos^2\alpha$$

To estimate the magnitudes of Δx and Δy the coordinate rotation must again be examined. In the following it is assumed that $\cos i \approx 1$ and $\sin i \approx i$ so that

$$\begin{bmatrix} x \\ y \end{bmatrix} = \begin{bmatrix} \cos(\omega+\Omega) & -\sin(\omega+\Omega) & i\sin\Omega \\ \sin(\omega+\Omega) & \cos(\omega+\Omega) & -i\cos\Omega \end{bmatrix} \begin{bmatrix} A \\ \sin v\cos A \\ \cos v\cos A \end{bmatrix}$$

The angle v is that caused by an error in the phase angle or time. If this error is of the order of 15 minutes or one degree then $\cos v \approx 1$ and $\sin v \approx v$ and the errors become

$$\Delta x = \Delta A\cos(\omega+\Omega) + vA\sin(\omega+\Omega)\Delta A - iA\sin\Omega\Delta A$$
$$\Delta y = \Delta A\sin(\omega+\Omega) - vA\cos(\omega+\Omega)\Delta A + iA\cos\Omega\Delta A$$

Since vA and iA are small quantities the errors in x and y can be represented in a worst case as

$$\Delta x = \Delta A$$
$$\Delta y = \Delta A$$

Since

$$\cos^2\alpha = \frac{x^2}{x^2+y^2}$$

The variance of the right ascension becomes

$$\sigma_\alpha^2 = \frac{y^2}{\left(x^2+y^2\right)^2}\sigma_x^2 + \frac{x^2}{\left(x^2+y^2\right)^2}\sigma_y^2$$

Using the approximation for σ_x^2 and σ_y^2 the above becomes

$$\sigma_\alpha^2 = \frac{\sigma_A^2}{x^2+y^2} = \frac{\sigma_A^2}{\cos^2\delta}$$

or

$$\sigma_\alpha = \frac{\sigma_A}{\cos\delta}$$

Using $\delta = 89^\circ$ and $\sigma_A = 0.01$ degree

$$\sigma_\alpha = \frac{0.01}{0.01745} = 0.573 \text{ degrees}$$

Low Thrust Maneuver Analysis

To more accurately predict the effect of a low thrust maneuver on the ATS-B orbit, as well as to provide a backup technique for inferring the maneuver velocity change $\overline{\Delta V}$ from knowledge of pre- and post-maneuver orbital elements, a preliminary analysis was made to determine the (nonimpulsive) orbit changes which result from a low thrust maneuver over a short time interval. It is to be expected that results should not differ appreciable from an impulsive $\overline{\Delta V}$ approximation, and this indeed is one conclusion of the study. It is also shown here, however, that because of an almost complete decoupling between thrusting and orbital motion which exists at synchronous altitude, the position change which accompanies a finite time maneuver can be very simply taken into account.

The following paragraphs summarize the analysis performed and make the following assumptions:

1. The satellite orbit is circular at synchronous altitude.
2. Earth oblateness is neglected.

3. The thrust acceleration $\bar{\epsilon}$ (thrust permit mass) is constant in magnitude and inertial direction throughout the maneuver time t.

In practice, the thrust is actually applied by a sequence of n small radial jet impulses at a fixed jet-start angle relative to the sun direction. Each impulse consists of a jet-on time corresponding to one-eighth of a spin revolution, and gives rise to a small effective velocity increment $\bar{\delta}$ aligned with the thrust direction at the midpoint of the jet-on time. Therefore, each $\bar{\delta}$ occurs in the same inertial direction. The equivalent continuous constant thrust acceleration $\bar{\epsilon}$ is taken to be that which would produce, over the maneuver time t, the same total velocity increment $n\bar{\delta}$ as n impulses over this time. Thus

$$\bar{\epsilon} t = n\bar{\delta}; \quad \bar{\epsilon} = \frac{n}{t}\bar{\delta} = \frac{1}{P}\bar{\delta} \tag{1}$$

where P is the satellite spin period (so that the maneuver time t corresponding to n impulses is t = nP).

Linearization

The effect of the above small constant thrust acceleration $\bar{\epsilon}$ on a synchronous circular orbit can be well approximated by a linearization about the unperturbed (nonthrusting) orbital position vector $\bar{x}$. Letting $\bar{y}$ represent the unperturbed (thrusting) orbital position and

$$\overline{\Delta x} = \bar{y} - \bar{x} \tag{2}$$

the (small) vector position difference, the differential equations of $\bar{x}$ and $\bar{y}$ motion are given by

$$\ddot{\bar{x}} = -\frac{\mu}{|\bar{x}|^3}\bar{x} \tag{3}$$

$$\ddot{\bar{y}} = -\frac{\mu}{|\bar{y}|^3}\bar{y} + \bar{\epsilon} \tag{4}$$

Where μ is earth gravitation constant. Replacing $\bar{y}$ by $\bar{x} + \overline{\Delta x}$ in Equation (4), and subtracting (3), there is obtained the differential equation of $\overline{\Delta x}$ motion.

$$\ddot{\overline{\Delta x}} = -\frac{\mu}{|\bar{x}|^3}\left[\frac{|\bar{x}|^3}{|\bar{x}+\overline{\Delta x}|^3}(\bar{x}+\overline{\Delta x}) - \bar{x}\right] + \bar{\epsilon} \tag{5}$$

Now using the law of cosines and retaining only linear terms in $|\overline{\Delta x}|$,

$$|x+\overline{\Delta x}|^{-3} = \left(|\bar{x}+\overline{\Delta x}|^2\right)^{-3/2} = \left(|\bar{x}|^2 + |\overline{\Delta x}|^2 + 2\bar{x}\cdot\overline{\Delta x}\right)^{-3/2}$$

$$\cong |\bar{x}|^{-3}\left(1 - \frac{3\bar{x}\cdot\overline{\Delta x}}{|\bar{x}|}\right) \tag{6}$$

Substituting (6) into (5) and retaining only linear terms in $|\overline{\Delta x}|$, the linear differential equation of $\overline{\Delta x}$ motion is found to be

$$\ddot{\overline{\Delta x}} = -\frac{\mu}{|\bar{x}|^3}\left(\overline{\Delta x} - \frac{3\bar{x}\cdot\overline{\Delta x}}{|\bar{x}|^2}\bar{x}\right) + \bar{\epsilon} \tag{7}$$

First Approximation

It is of interest at this point to compare the relative magnitudes of the two terms on the right side of Equation (7). The jet thrust, averaged over one impulse on-time (one-eighth of a spin revolution), is typically about three pounds (considered to act along the on-time midpoint direction). Taking spacecraft weight as about 750 pounds = 23.3 slugs gives a velocity increment per impulse of magnitude

$$\delta = \frac{3}{23.3} \times \frac{P}{8}, \text{ ft/sec} \tag{8}$$

and from (1) a constant thrust acceleration of magnitude

$$\epsilon = \frac{3}{23.3} \times \frac{1}{8} = 1.6 \times 10^{-2} \text{ ft/sec}^2 \tag{9}$$

On the other hand, the first term on the right of (7) can never exceed

$$\frac{\mu}{|\bar{x}|^3}\,|\overline{\Delta x} + 3\overline{\Delta x}| \tag{10}$$

in magnitude. From the type of maneuvers performed in actual operations, it is known that the initial post-maneuver displacement $|\overline{\Delta x}|$ from the corresponding unperturbed orbital position is never greater than about two nm. For this $|\overline{\Delta x}|$ and synchronous radius $|\overline{x}|$, expression (10) is evaluated to be 2.6×10^{-4} ft/sec^2, which is only about 1.6 percent of the ϵ contribution (9). Thus, to a good first approximation, Equation (7) becomes

$$\ddot{\overline{\Delta x}} = \overline{\epsilon} \tag{11}$$

Equation (11) has immediate solutions for perturbed position and velocity vectors at the end of maneuver time t:

$$\overline{\Delta x} = \frac{t^2}{2}\overline{\epsilon} \tag{12}$$

$$\dot{\overline{\Delta x}} = t\epsilon \tag{13}$$

where t = 0 refers to maneuver start time, and where these solutions are particularized to the initial conditions $\overline{\Delta x} = \dot{\overline{\Delta x}} = 0$ (the perturbed orbit state initially coincides with that of the unperturbed orbit).

Since the form (11) neglects a small contributing acceleration term, it follows that solutions (12-13) may be expected to be accurate only over some limited range of t. It is in fact shown below that for maneuver times of interest, the neglected acceleration does not contribute appreciable to the first approximation solutions (12-13).

Equations (12-13) state that to find the orbital state at the end time t of the maneuver, one merely adds to the unperturbed position and velocity vectors (at that time) certain corrections $\overline{\Delta x}$, $\dot{\overline{\Delta x}}$, which lie entirely along the constant direction of the applied thrust acceleration $\overline{\epsilon}$. This procedure is shown pictorially on an enlarged scale in Figure 6 for $\overline{\epsilon}$ aligned with the initial tangential and radial directions of the unperturbed orbit. The simplicity of (12-13) arises from the essential "de-coupling" of thrusting and orbital motion which exists at the large synchronous radius.

To relate (12-13) to an impulsive maneuver approximation, it is seen that the velocity change $\dot{\overline{\Delta x}}$ is exactly equivalent to the $\overline{\Delta V}$ accruing from constant thrust acceleration $\overline{\epsilon}$ over the maneuver time t, where this $\overline{\Delta V}$ is considered to be instantly applied at time t on the unperturbed orbit. However, the pre- and post-maneuver orbits do not agree in position at t: a displacement $\overline{\Delta x}$ given by (12) exists. In practice (see examples below) this displacement is fairly small, and so to this extent the approximation of an impulsive maneuver applied at time t is a reasonable one. But the simplicity of the nonimpulsive expressions (12-13), which is also preserved in refinements of these given below, suggests that the position displacement may just as well always be included in maneuver prediction and evaluation work.

Second Approximation

A more exact solution of (7) which includes the small effect of the first term can be developed by a power series expansion for $\overline{\Delta x}$ which retains the first contributing term beyond t^2. Thus, a solution is assumed of the form

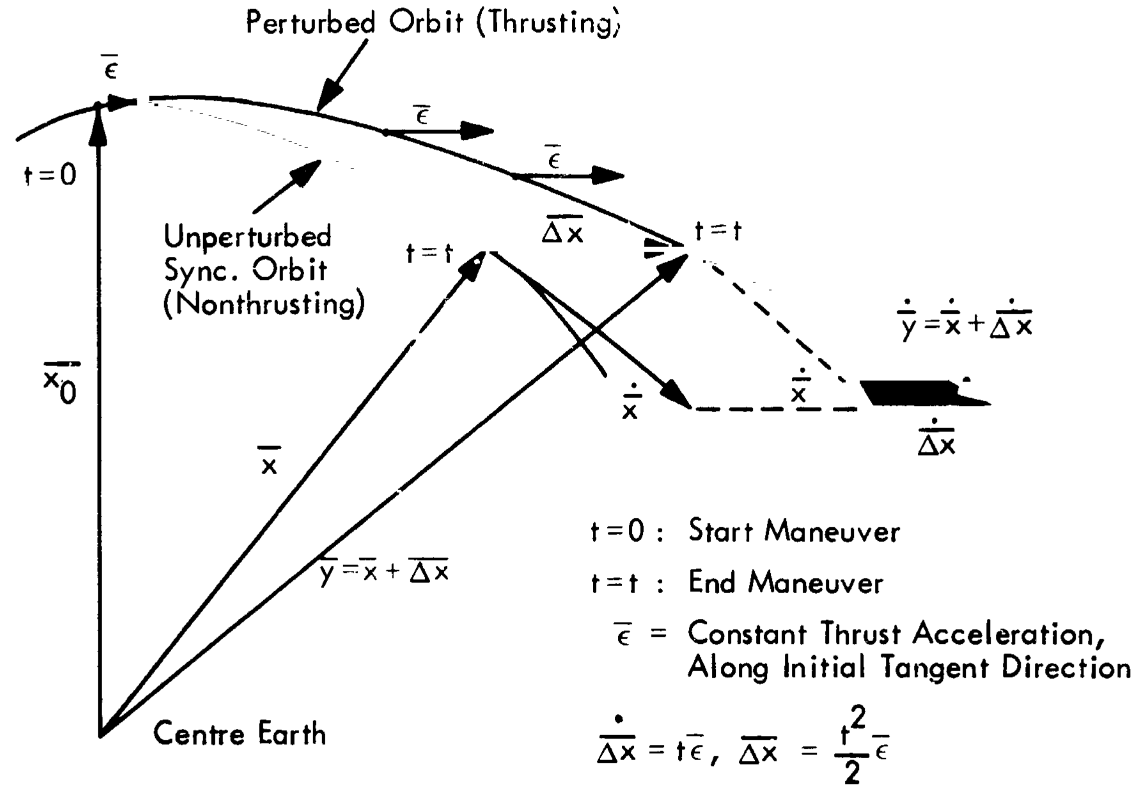
$$\overline{\Delta x} = \overline{a}_o + t\overline{a}_1 + t^2\overline{a}_2 + t^3\overline{a}_3 + t^4\overline{a}_4 + \ldots \tag{14}$$

where in fact $\overline{a}_o = \Delta\overline{x}\,(0) = 0$ and $\overline{a}_1 = \dot{\overline{\Delta x}}\,(0) = 0$, since the perturbed orbit coincides in position and velocity with the unperturbed orbit at the start of the maneuver. The unperturbed orbit $\overline{x}$ is also assumed to adequately represented over the maneuver time by the first few terms of a power series,

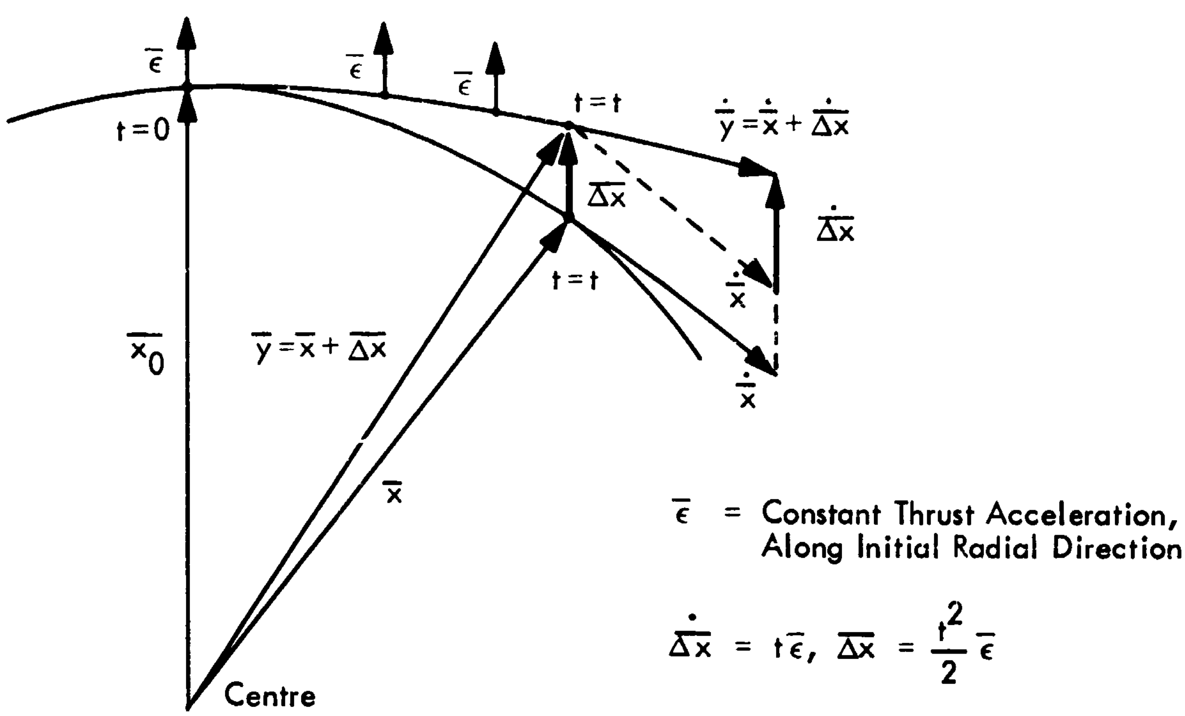
$$\overline{x} = \overline{x}_o + t\dot{\overline{x}}_o + \frac{t^2}{2}\ddot{\overline{x}}_o + \ldots \tag{15}$$

where the vector coefficients in this series are known. The magnitude $|x|$ is of course constant for the assumed circular unperturbed orbit. After substitution of (14) and (15), (7) becomes

$$\begin{aligned} 2\overline{a}_2 + 6t\overline{a}_3 + 12t^2\overline{a}_4 + \ldots = &-\frac{\mu}{|x|^3}\left(t^2\overline{a}_2 + t^3\overline{a}_3 + t^4\overline{a}_4 + \ldots\right) \\ &+ \frac{3\mu}{|x|^5}\left[t^2\left(\overline{a}_2\cdot\overline{x}_o\right) + t^3\left(\overline{a}_2\cdot\dot{\overline{x}}_o + \overline{a}_3\cdot\overline{x}_o\right) + \ldots\right] \\ &\left(\overline{x}_o + t\dot{\overline{x}}_o + \frac{t^2}{2}\ddot{\overline{x}}_o + \ldots\right) + \overline{\epsilon} \end{aligned} \tag{16}$$



a) Tangential Thrust



b) Radial Thrust

Figure 6. Orbital State Changes Resulting from Low Thrust Maneuvers

Equating coefficients of like powers of t up to t^2 in (16), there results

$$2\bar{a}_2 = \bar{\epsilon}\,; \quad \bar{a}_2 = \frac{\bar{\epsilon}}{2}$$

$$6\bar{a}_3 = 0$$

$$12\bar{a}_4 = -\frac{\mu}{|\bar{x}|^3}\bar{a}_2 + \frac{3\mu}{|\bar{x}|^5}\left(\bar{a}_2 \cdot \bar{x}_o\right)\bar{x}_o\,; \quad \bar{a}_4 = -\frac{\mu}{24|\bar{x}|^3}\bar{\epsilon} + \frac{\mu}{8|\bar{x}|^5}\left(\bar{\epsilon} \cdot \bar{x}_o\right)\bar{x}_o \tag{17}$$

With these coefficients substituted in (14), the perturbed position and velocity vectors take the approximate form

$$\overline{\Delta x} = \frac{t^2}{2}\bar{\epsilon} - \frac{\mu}{24|\bar{x}|^3}t^4\left[\bar{\epsilon} - \frac{3\;\bar{\epsilon}\cdot\bar{x}_o}{|\bar{x}_o|^2}\bar{x}_o\right] \tag{18}$$

$$\dot{\overline{\Delta x}} = t\bar{\epsilon} - \frac{\mu}{6|\bar{x}|^3}t^3\left[\bar{\epsilon} - \frac{3\;\bar{\epsilon}\cdot\bar{x}_o}{|\bar{x}_o|^2}\bar{x}_o\right] \tag{19}$$

The first terms of which are seen to be in agreement with (12-13).

The solutions (18-19), or the more approximate forms (12-13), are useful in two ways:

1. To predict the post-maneuver orbital state $\bar{y} = \bar{x} + \overline{\Delta x}$, $\dot{\bar{y}} = \dot{\bar{x}} + \dot{\overline{\Delta x}}$ resulting from application of a known (magnitude and direction) thrust acceleration $\bar{\epsilon}$ over maneuver time t.

2. To evaluate, from knowledge of pre- and post-maneuver orbital states at time t, the thrust acceleration vector $\bar{\epsilon}$ which caused the orbital change. This can be accomplished by solving either (18) or (19) (or (12) or (13)) for $\bar{\epsilon}$. According to whether the post-maneuver orbital position or velocity is more accurately known, the choice would fall to (18) or (19), respectively. Alternately, both equations can be solved for $\bar{\epsilon}$ and the results compared, averaged, etc.

Examples

To illustrate application (1) above and give some indication of the maximum size of the numbers involved, consider the orbital state change resulting from $\bar{\epsilon}$

of magnitude (9), aligned with the initial tangential direction of the synchronous orbit (Figure 6), and applied for an on-time of $t = 20$ min $= 1200$ sec. This is about the maximum maneuver time encountered in practice. For this case, $\bar{\epsilon} \cdot \bar{x}_o = 0$, so that the second term in the brackets of (18) and (19) vanishes. The position and velocity increments then lie entirely along the initial tangential direction of $\bar{\epsilon}$, and are evaluated in magnitude as

$$\Delta x = 11,500 \text{ ft} -7.4 \text{ ft}$$

$$\Delta \dot{x} = 19.2 \text{ ft/sec} -0.024 \text{ ft/sec} \tag{20}$$

It can be seen from these numbers that the first terms of (18) and (19) overwhelmingly dominate the final solution.

As a second example, suppose $\bar{\epsilon}$ to be applied along the initial outward radial direction of the synchronous orbit. In this case $(\bar{\epsilon} \cdot \bar{x}_o) = \bar{\epsilon} \, |\bar{x}_o|$, and $\bar{x}_o = \frac{|\bar{x}_o|}{\bar{\epsilon}} \bar{\epsilon}$. The bracketed terms in (18) and (19) thus reduce to $\bar{\epsilon} - 3\bar{\epsilon} = -2\bar{\epsilon}$, and the position and velocity increments lie entirely along the initial radial direction of $\bar{\epsilon}$. For the magnitude (9) of $\bar{\epsilon}$, the perturbations are found to be

$$\Delta x = 11,500 \text{ ft} + 14.8 \text{ ft}$$

$$\Delta \dot{x} = 19.2 \text{ ft/sec} + 0.048 \text{ ft/sec} \tag{21}$$

where again the first terms are by far the major contributors.

Discontinuous Thrust Consequences

A final point of interest concerns the error incurred in the preceding analysis by the assumption that the thrust acceleration $\bar{\epsilon}$ is constant in magnitude. For this purpose it is convenient to deal with the first approximation solutions (12-13), since as shown above they are valid approximations for the maneuver times of interest. It is recalled that the actual thrust time history consists of a sequence of n short impulses, each of which occupies an interval of time P/8 (P = spin period), with separation interval between successive pulses of 7P/8. Each impulse gives rise to a velocity increment $\bar{\delta}$. It is now assumed that this thrust history is well approximated by a sequence of equal "true impulsive" velocity increments $\overline{\delta}$, each occupying zero time of itself and separated from the succeeding member of the sequence by a full spin period P. With this additional assumption, it is clear that no error in total velocity increment $\dot{\overline{\Delta x}}$ is incurred by the use of (13), with the constant thrust acceleration level $\overline{\epsilon}$ defined as in (1), since

$$\dot{\overline{\Delta x}} = n\overline{\delta} = n\left(\frac{t}{n}\overline{\epsilon}\right) = t\overline{\epsilon} \tag{22}$$

which agrees exactly with (13).

The actual relative position displacement $\overline{\Delta x}$ (taken as uncoupled from orbital motion) due to a sequence of n impulsive velocity increments $\overline{\delta}$ separated in time by a spin period P is

$$\overline{\Delta x} = \overline{\delta}(nP) + \overline{\delta}\left[(n-1)\ P\right] + \ldots + \overline{\delta}(2P) + \overline{\delta}(P) \tag{23}$$

That is, the effect of the first $\overline{\delta}$ extends over n spin periods, . . . , and the effect of the nth $\overline{\delta}$ extends over one spin period. Equation (23) simplifies to

$$\begin{aligned}\overline{\Delta x} &= \left[n + (n-1) + \ldots + 2 + 1\right] P\overline{\delta} = \frac{n}{2}(1+n)\ P\overline{\delta} \\ &= \frac{n^2P\overline{\delta}}{2}\left(1+\frac{1}{n}\right)\end{aligned} \tag{24}$$

But from (1) and the relation $t = nP$, where t is total maneuver time corresponding to n impulses, it follows that

$$n^2P\overline{\delta} = nt\overline{\delta} = t^2\overline{\epsilon} \tag{25}$$

so that (24) becomes

$$\overline{\Delta x} = \frac{t^2}{2}\left(1+\frac{1}{n}\right)\overline{\epsilon} \tag{26}$$

which agrees with (12) except for the factor $\left(1+\frac{1}{n}\right)$. Now from (8), with a spin period for ATS-B of about 100 rpm (so that P = 0.6 sec), the velocity increment per impulse is about 1/100 ft/sec. Typical maneuvers are never less than about 1 ft/sec total $\overline{\Delta V}$, so that n is never less than 100. Thus, in (26) the factor $\left(1 + \frac{1}{n}\right)$ is always close to unity, and the form (12) is recovered. It is concluded that the assumption of continuous thrust acceleration specified by (1) introduces negligible error into the first approximation solutions (12-13).

CONCLUSIONS AND RECOMMENDATIONS

Based on analysis results detailed in Reference 1, the following conclusions and recommendations regarding the successful December ATS launch are submitted.

Conclusions

- For the nominal ATS transfer orbit, ignition at first apogee is limited primarily by attitude determination and station coverage considerations; spacecraft orbit determination is not a limiting factor.

- With current attitude data transmission and processing procedures, first apogee ignition is possible only when a prior reorientation maneuver is not required and only if Toowoomba command capability is not lost at the low elevation angles existing at first apogee.

- For the December launch, apogee motor ignition at second apogee was fully successful and yielded results which proved to be closer to desired values than predicted. The importance of attitude verification before apogee motor ignition was emphasized by the results of subsequent spacecraft attitude maneuvers.

- Program ATTDET successfully determined spin axis attitude during the transfer orbit and reorientation maneuver phase, using combined POLANG and sun data. Following final erection, attitude solutions have been largely based on ACO sun data. Weights (error standard deviations) of 1.0 and 0.15 degrees for POLANG and sun data, respectively, have been generally employed throughout the mission.

- From comparison with transfer orbit attitude based on an independent solution, and from analysis of ATTDET solutions employing different station/data mixes, it is estimated that the accuracy of attitude determination throughout the mission has been on the order of 0.5 to 1.0 degrees (great circle arc). Improvement depends on reduction of POLANG measurement errors, determination of possible POLANG biases (by station), and resolution of differences in the two types of sun data available.

- POLANG data from all stations was smoothest during the transfer orbit, with quality declining after the first erection maneuver; only irregular readings have been received since final erection. Random error was usually on the order of 0.5 to 1.0 degrees, increasing at times to about 2.0 degrees. Evidently incorrect readings between 3 and 10 degrees different from expected or trend values have been occasionally received from all stations. Manual editing based on prediction curves has been successfully applied to detect and remove such points.

- Sun data from ACO and SC sources has continually differed by 0.2 to 0.5 degrees throughout the mission, with no conclusive evidence to date on which is the more accurate. SC data has been available only intermittently since spacecraft arrival on-station. Determination of sun data errors will permit more meaningful assignment of data weights in the ATTDET program.

- ACO sun data as received at the computer is specified only to the nearest 0.1 degree, with significance equivalent to SC data being available only when an incremental 0.1 degree change is observed in the readings.

- POLANG bias cannot be reliably estimated from single-station data alone, but must be inferred from differences between attitude solutions obtained using data from a known-unbiased station alone, and data from that same station plus another (possibly biased) station.

- Preliminary postflight analysis indicates Rosman and Mojave were probably unbiased during the transfer orbit, with Toowoomba and Kashima bias estimated as approximately -0.7 and +0.5 degrees, respectively. Additional testing is needed to more firmly establish these levels as well as individual station biases at other times during the mission.

- POLANG bias from a given station may change at different times during the mission, because of changing calibration and/or Faraday rotation correction errors.

- Sun data alone was incapable of fixing attitude during the transfer orbit, owing to the relatively short time interval involved. Since arrival on-station, attitude obtained from ACO data alone over periods of 4 to 8 days has closely agreed with solutions incorporating available POLANG data.

- A method has been developed to determine ATS-1 spin axis from on-board Soumi camera pictures. An error analysis based upon conservative assumptions indicates that spin axis declination can be determined to within 0.01 degrees. Right ascension accuracy depends upon declination and becomes indeterminate when $\delta = 90$ degrees; for $\delta = 89$ degrees, however, right ascension should be determinable to within 0.6 degrees.

Recommendations*

- If ignition at first apogee is a firm requirement for future ATS missions, it is recommended that an apogee motor ignition command capability be assigned to the Kashima station. Alternatively, a short duration ignition timer (which may be actuated by Toowoomba shortly before first apogee) may be incorporated into the spacecraft. Means to expedite the transmission and processing of early attitude data are also required to ensure ignition at first apogee.

- The bias estimation mode of program ATTDET should be sparingly employed, and then only when data from two stations are available, with substantial prior assurance (from the data history) that one station is unbiased. This serves to fix a reference level for the data, against which any bias from the other station can be estimated.

- Whenever high-level POLANG noise is present, manual editing is recommended, as e.g., by comparison of predicted time history trends based on previous attitude estimates. Compression of ten-point POLANG messages into single averaged points ("distilled" data) is also recommended as a means of significantly reducing POLANG data volume to ATTDET without appreciably affecting the solution estimates.

- Investigation should be undertaken to determine the causes of high random noise, isolated large-bias readings, and altogether unuseable blocks of points which have been encountered in POLANG data at various times through the mission.

- Because of oscillatory behavior sometimes experienced in the reweighting mode of ATTDET, it is recommended that the solution after first convergence of the iterative loop (reweighted mode solution disregarded) be regarded as the preferred attitude estimate, especially if data weights are based on previously determined RMS's of residuals.

- Additional solution testing should be undertaken with an aim toward evaluating the relative accuracy of sun data as given by the ACO and synchronous controller readings. If the question can be successfully resolved, only the preferred value (when both are available) should henceforth be employed in program runs.

- For on-station attitude near the south pole and a near-synchronous orbit, POLANG should vary sinusoidally at orbital rate and with amplitude proportional to declination increment away from $\delta = -90$ degrees. Accordingly, attitude determination would be enhanced if POLANG data from at least one station could be gathered at regular intervals over a complete 24-hour orbital period.

*Some of these recommendations also appear in QPR-2 (Ref 2) and are repeated here to complete the summary of Reference 1.

- Comparison tests of ATTDET solutions obtained using single-station data during various phases of the mission should be continued in order to isolate possible POLANG biases, as well as to determine solution sensitivities to calibration and Faraday rotation correction errors.

- To improve spacecraft attitude control procedures the following items, revealed during real-time performance of reorientation maneuvers, should be investigated for feasibility prior to the next ATS launch:

    1. Methods to automatically generate the reorientation plot used for real-time monitoring of attitude maneuvers.

    2. Methods to reduce the time dependence of the reorientation plots used in real-time monitoring.

    3. Application of other attitude sensors for attitude determination at synchronous altitude, with a view towards flying an attitude determination package on a future ATS.

    4. Study of noise in real-time POLANG data with a view toward improving the reorientation touch-up procedure and determining the sources of this noise.

    5. Use of an accurate monitor chart during attitude changes so as to eliminate these charts as an error source.

Results of the ATS-B velocity correction maneuvers indicated several areas where additional procedures or techniques would help in formulating the required decisions:

- At present, spacecraft position is determined by using a mathematical model of its orbital dynamics, i.e., via some orbit prediction method. The accuracy of such models depends upon the inclusion of all important forces, the accuracy of the various constants involved, the integration method employed, etc. To supplement the present method, it is suggested that a method to determine spacecraft position by means of range measurements be developed. This geometrical method of fixing spacecraft position entails solving for the intersection of (at least) three known spheres. A sensitivity or error analysis should also be performed to determine if such a method can yield reasonable position tolerances. The objective is to provide an independent check or comparison against predicted orbital position when the spacecraft is on-station and drifting very slowly.

- At present there is no accurate means of estimating the direction in which velocity corrections are actually generated, except when the corrections exceed 50 fps in magnitude. It is recommended that some effort be directed to the use of the orbital element impulse equations to resolve this question. This could be in the form of a simple summation where the effects of each pulse (considered as an impulse) are considered individually on each orbital element or by some more elaborate numerical integration method. The pulse definitions and directions of application would then be adjusted so that the initial elements would hopefully agree with the final elements. The results could also be used for more realistic maneuver planning.

- To enhance the ability to measure in-orbit velocity corrections and calibrate the peroxide systems it is recommended that a study be made to determine if on-board accelerometers can be utilized to provide additional information.

- The present pressure measurements are sensitive to 1.5 psia. When small corrections are made, it is impossible to use pressure as a monitor of system performance. It is suggested that a study be made to see if a more sensitive pressure sensor can be used.

- The present peroxide system calibration curves allow for a maximum velocity correction of 15 fps at full pressure. It is recommended that these curves be extended before the next launch so as to include maneuvers of up to 50 fps.

- At present some of the peroxide system parameters (e.g., rise times) are given for only one pressure. Since the actual pressure rapidly departs from this value, it is suggested that several intermediate values be given. This is motivated by the need for as accurate an engine calibration as possible for the required "fine-grain" on-station maneuvers.

- More accurate or useful information would be desirable when the number of pulses drops below 50. It is expected that some maneuvers will be called for in both the area of reorientation and velocity correction where the pulse requirement will be below 50.

PROGRAM FOR NEXT REPORTING INTERVAL

During the next reporting interval, the picture-attitude method described herein will be programmed and tested with pictures from 7 January 1967 . This data was chosen because an independent group working with the Weather Bureau has determined the spin axis declination at this time. Once the program is tested, attitude will be found on a continuing basis to see if any noticeable changes are occurring.

Task 5

ATTITUDE SYSTEM DESIGN

DISCUSSION

Phase I of the Attitude System Design task plan, "Task Description of Initial System/360 Conversion Effort" has been completed for three major program systems, TOS/ESSA Attitude Determination System, Prelaunch and Utilities Programs, and Multi-Application Subroutines. Phase II which involves the actual conversion of the program to run on the System/360, was started. At the end of Phase II each program being converted should produce output on the System/360 in agreement with bench marks established on the 7094.

The three major program systems contain 36 separate programs or generalized subroutines that are included in the conversion effort. This is one additional program from the last reporting interval with the addition of the program to merge ORB1 tapes. The following describes the status of the System/360 conversion:

	No. of Programs	Percent
Total number of programs and subroutines being converted	36	100
Programs completed, Phase I	36	100
Programs completed, Phase II	23	64

The TOS/ESSA Attitude Determination System presented a major conversion problem in the development of a method for reading a raw digital data tape. A test program was written using the EXCP macro instruction to read the tape. The raw digital data tape has no record gaps and there is not enough room in memory to read in the whole tape at one time. Therefore, it must be read using a multi-buffer technique. As a buffer becomes full, it is written on an intermediate tape, thus freeing the buffer for more data.

Only two input buffers were used during the first attempts to read the tape. Since all input/output operations are privileged, the problem program must call Operating System/360 to have these operations performed. Once the system has control it performs an input/output request and any other housekeeping chores for that task or other tasks that it deems necessary. This makes timing of the time lapse impossible during the interval when the system has control. With only two buffers it was discovered that there was a data loss due to this timing problem. The solution was to modify the test program to read into four input buffers. This program will now read a gapless raw digital data tape and generate an output tape in record format.

A System/360 version of PREPRO, the program to read a raw digital data tape, has been written to incorporate the logic from the test program (i.e., to generate an intermediate tape with a standard record format that can then be processed normally). This program is being debugged. PREPRO edits and reformats the data from the raw digital data tape for other programs in the system.

The other major programs in the TOS/ESSA Attitude Determination are all in Phase II of the conversion effort. Results from the System/360 version of ALS/ARS are nearly in agreement with the bench marks established on the 7094. Further test runs are being made to reduce the differences in the output and to ensure proper operation of the program in all cases. The preliminary, necessary tape conversions to provide input to DATPRO, JIFFY and STADEE have been completed and debugging runs have started. Two of the TOS/ESSA programs, JCAL and ASP-MGAP, are completed through Phase II.

The second major program system being converted consists of the Prelaunch and Utility Programs (PUP) program system. Most of these programs have been converted to System/360 operation and agreement with bench marks was achieved. These programs WMSAD, WMSUM, GAMWIN, QUIKEL and CORFOE are being subjected to additional tests to ensure that the programs have been properly converted. One additional program, O1MERG, has been added to the conversion effort for PUP. This program, which was a 7094 assembly language program, is being completely rewritten for the System/360.

The third major programming area being converted is Multi-Application Subroutines (MAS). All of the generalized subroutines in the MAS program system have been converted to System/360 operation. TAPRE, the subroutine used to read the ORB1 tape is still undergoing tests to evaluate the slight discrepancies encountered in 7094 comparison runs.

The programs already converted to the System/360 are being run in parallel with 7094 operation runs to allow a thorough checkout of the programs and provide a basis for operational acceptance of these programs. To facilitate the operational checkout of these programs and to eliminate deck handling, a system tape containing the programs was generated.

PROGRAM FOR NEXT REPORTING INTERVAL

Phase II of the task plan will be completed and a thorough checkout will be made of the programs on the System/360. After each individual program has been converted and tested the complete program systems will be run as a unit to ensure compatibility and to eliminate other communication problems. It will also be necessary to replace the operational support areas (such as tape libraries) that currently exist on the 7094.

CONCLUSIONS AND RECOMMENDATIONS

Phase I of the conversion effort has been completed and Phase II, the actual conversion of programs to operate on the System/360, is underway. Recent discussions on the Attitude System Design task have revealed that delivering a group of converted programs is not sufficient to establish operations on the System/360. In addition to testing the individual programs and program systems, it will be necessary to establish the support apparatus such as tape libraries, system tapes, etc., to run operationally. As a result, it will be necessary to move the completion date on Phase II conversion to an earlier date to allow time for the operational support apparatus to be established.

Task 6

AE-B

DISCUSSION

Daily Attitude

Attitude results from real-time raw digital aspect tapes continued to be processed on a daily basis. To maintain mission attitude, operational support was provided to the project office: two spin-downs, one spin-up sequence, and three steering commands. Spin rate determination using optical and sun slit data indicate the spacecraft spin rate decayed from 6.5 rpm on orbit 3125 to 0.0 rpm on orbit 3137, and then an increase to 8.0 rpm on orbit 3147. This decrease to zero and subsequent increase indicates the spacecraft is now spinning in the opposite direction. The latest phimax (maximum roll angle) results are approximately 37 degrees from negative orbit normal and the spin rate is 37 rpm and increasing.

Definitive Attitude

Final definitive cards and graphs for orbits 600-1980 have been delivered to the AE-B Project Office, completing definitive work through 1 November 1966. Definitive attitude and standard deviations for these orbits are as follows:

Orbits	Standard Deviation
600 - 696	1.13
697 - 975	6.22
976 - 1123	3.44
1124 - 1234	6.95
1235 - 1619	1.71
1620 - 1992	1.85

The high standard deviations in orbits 697 through 1234 were caused by a scatter in the magnetometer data which resulted when the gamma angle was outside of the DSAI limits. In addition, a minimal amount of optical data was available due to terminator interference.

Analysis and Programming

Several programming improvements and changes were made to the AE-B system. These modifications resulted from analyses performed to correct the problems that occurred in determining the attitude of the AE-B spacecraft. The problems encountered and the resulting analyses and programming changes are discussed in the following paragraphs.

Spin-Down to Zero Rpm

During a spin-down on 6 January a radical decrease from 25 rpm to 12 rpm within a six-hour period was noticed in the spin period. The AE-B Control Center was advised of this and indicated that they also noticed the jump and attributed it to gas leaking from the spacecraft. Subsequently, the spacecraft spun down to 0.0 rpm (about orbit 3137) and then spun up to approximately 8 rpm at orbit 3147. This last increase was assumed to be in the negative direction, which led to a study on the effects on the AE-B processing of a spacecraft spinning in the opposite direction. First, the following programs were studied:

- AEPROC—changes were made to handle a negative spin rate and the modified program is currently operational.
- AEB-MGAP—this program was tested for an opposite spin (i.e, negative spin rate) with no spin torque. It correctly computed the spin rate decay, tending toward a spin rate of zero. It also predicted a spin axis drift in the opposite direction to that computed for the positive spin rate. The command search option was tested and commands were generated to drive the spin axis toward the orbit normal.

Three problems were anticipated as a result of the negative spin rate:

1. The MGAP test revealed that under a negative spin, the spin axis will diverge from the orbit normal much more rapidly than under a positive spin. This effect is due to the fact that the negatively charged permanent magnet (designed to cause the spin axis to track the orbit normal

at a rate of about 2 degrees a day) will develop an opposite torque when the spin direction is reversed. The spin axis will then move approximately 2 degrees a day in the opposite direction from the orbit normal, thus moving away from orbit normal at approximately 4 degrees a day. A correction to the spin axis at least once every two days, and possibly more often, will be required to maintain the axis within 5 degrees of orbit normal.

2. The status of a negatively spinning satellite locked into a torque mode tending toward negative spin can be compared to a positively spinning satellite locked into spin-up mode. Because AEB-MGAP will predict only for a simple decay state with no applied torque, a short-term decay constant must be hand-computed periodically. This constant is input to MGAP until the spin rate approaches a steady state where the decay torque is equal to the applied torque.

3. MGAP does not predict attitude for a nutating satellite. It has not been possible to predict the amount of nutation which will occur when the satellite spin rate approaches zero. As the satellite spins up in the negative direction the amount of nutation should decrease. However, if the nutation is initially large, MGAP predictions will be unreliable while the satellite spin rate is slow.

Loss of DSAI Readings

During the spin-down mentioned above, spin rate determination was hampered because the DSAI could not see the sun. Since sun slit indications are used to automatically compute spin rates, it was necessary to compute spin rate from mid-scan times constructed from the optical sensor data. This was programmed in subroutine TIMSPN, which includes a correction for orbital angular velocity.

The intervening number of rotations and average spin period are computed for each pair of consecutive times as follows

let E = the magnitude of the expected spin period

T = the tolerance

D = any time difference which will yield an unambiguous result for the number of rotations between two observations

R = the number of rotations associated with D

then,

$$R(E+T) < (R+1)(E-T)$$

$$R < E-T/2T$$

and $D \leq (E+T)R$

Therefore,

$$D_{lim} = (E+T)(E-T)/2T$$

(If $D_{lim} = E+T$, D_{lim} is set to E+T, and R is set to 1)

All time differences greater than or equal to D_{lim} are rejected. Each remaining time difference is averaged over the intervening number of rotations, corrected for orbital angular velocity, and tested against the expected spin period and tolerance. Those values meeting the tolerance test are input to the sigma rejection routine to produce a smoothed spin rate.

The orbital correction is computed as follows

let $\bar{R}_1$ and $\bar{R}_2$ = the position vectors at the first and second observation times, respectively

$\bar{S}$ = the spin axis vector

$*$ = a unitized vector

e = the satellite's angular orbital motion in a plane parallel of rotation (spin)

then,

$$\cos e = (\bar{R}_1 \times \bar{S})^* \cdot (\bar{R}_2 \times \bar{S})^*$$

The true (corrected) spin period P_t is then computed from the observed spin period P_0 as follows

For the "forward-rolling" satellite (spin in same direction as orbital motion)

$$P_t = \frac{P_0}{1 + \frac{e}{2\pi}}$$

where e is in radians. For the "backward-rolling" satelling (spin opposing orbital motion)

$$P_t = \frac{P_0}{1 - \frac{e}{2\pi}}$$

For a more detailed discussion of this correction, see "Spin Rate Computation" in the DATPRO documentation for TOS/ESSA (Ref 3).

Fluctuations in Magnetometer Data

Recent difficulties encountered in obtaining consistent AE-B attitude solutions from magnetometer and sun data have prompt[illegible] examination of the quality of magnetometer data currently being processed. Data from orbits 3510 through 3582 have been reviewed and the attitude solutions compared, both with and without the addition of sun data. The following observations are relevant.

a. Between orbits 3510 and 3540, unexplained fluctuations have been observed in magnetometer data of such amplitude as to render the readings all but meaningless. Typically, the computed magnetometer angle between spin axis and field vector varies over a spread of 80 to 100 degrees during a given message taken from these orbits (see Figure 7). Moreover, the X-magnetometer readings change erratically over a wide range of both positive and negative values ($\pm$200 milligauss). Such data is useless for purposes of attitude determination, and should be discarded immediately if received in such form on subsequent orbits.

b. For orbits 3540 through 3582, fluctuations in magnetometer data and resulting magnetometer angle (corrected for previously established misalignments) appear greater than expected from random noise, but still may be of some use. Here, the raw data (X-magnetometer) is at least consistent in sign, and the spread in magnetometer angle is on the order of 15 to 20 degrees (Figure 7). Data of this quality may tentatively be retained for further processing. Manual screening of messages is a necessary interpretive step in the solution process although, in practice, rejection of solutions can be made after processing by CONES.

c. For data of the same quality type as 3540 through 3582 mentioned above, a further reasonableness test can be made by comparing magnetometer-only solutions with magnetometer plus sun angle solutions. This approach was studied for orbits 3450 through 3582, with the following results being representative of the two possibilities which can occur:

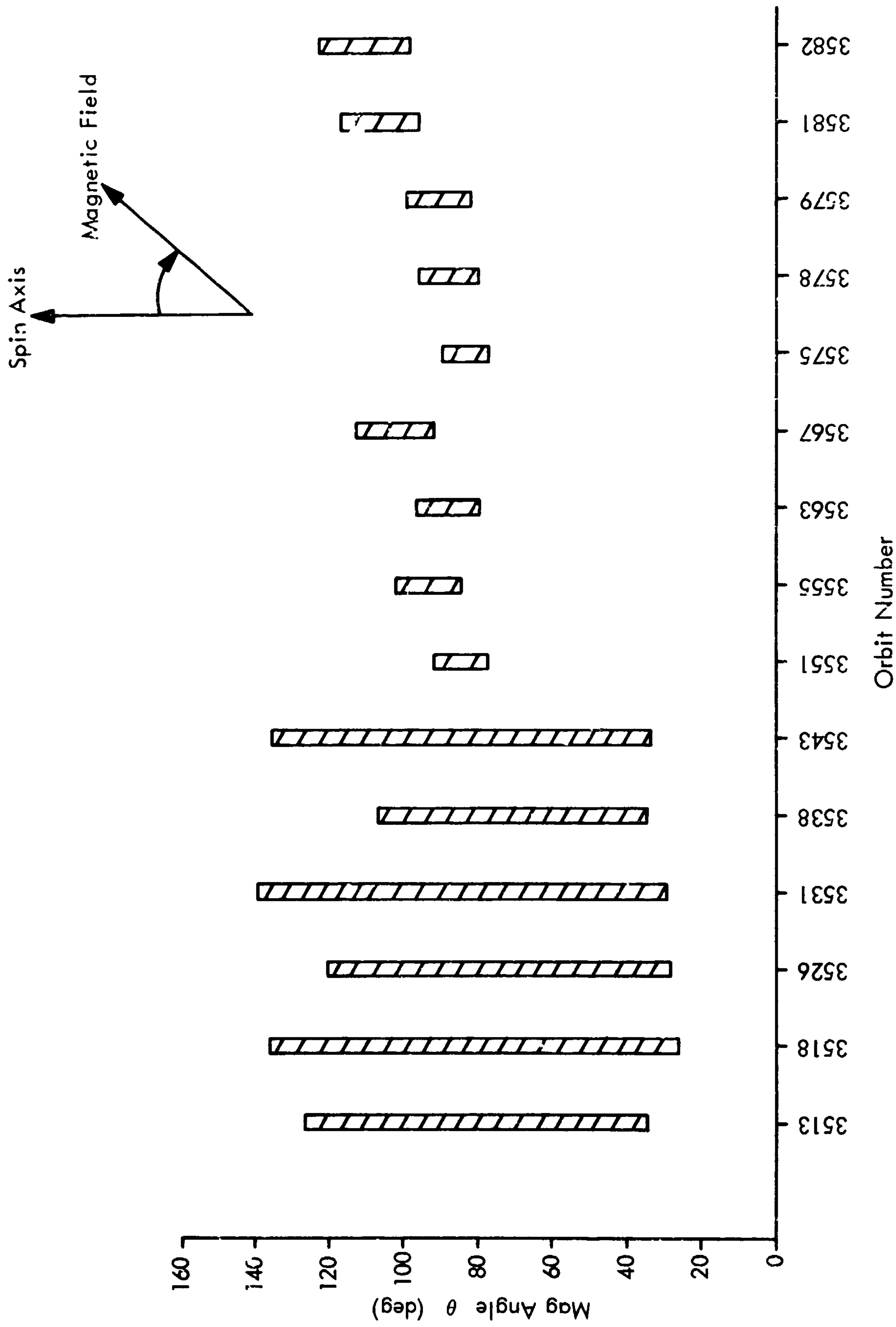


Figure 7. Computed Mag Angle Spread Over AE-B Message Interval

Orbit	α	δ	ϕ max	γ (computed)	γ (observed)	Data
3579	225.1	-24.2	26.2°	118.3°		Mag-only
3579	215.2	[illegible].2	17.3	124.1	125.3°	Mag-plus-sun
3581	214.9	-27.0	17.1	125.8		Mag-only
3581	215.4	-27.1	17.6	125.3	125.3	Mag-plus-sun

For orbit 3579, the magnetometer-only solution differs considerably from the magnetometer-plus-sun solution, while for orbit 3581 the two solutions are in good agreement. It may be concluded, then, that the 3579 magnetometer message was unreliable while that of 3581 was reasonably acceptable. In both cases the internal fluctuations of magnetometer data over the message were of comparable amplitude. Comparative solution tests of this nature are recommended for future runs employing "low" noise magnetometer data, by establishing greater confidence in the attitude results obtained from each data message.

PROGRAM FOR THE NEXT REPORTING INTERVAL

Since sun data remains of fairly good quality, an attitude solution based on sun data alone would apparently be of value. Several days must elapse, however, before a "good" intersection of sun cones can be achieved and, during this time, the attitude may not be taken as constant (as assumed in CONES). Nevertheless, it may be possible to deduce a solution if the model of α, δ motion with time is taken to be simply linear (an adequate assumption over a period of 5 to 10 days). Depending on the life expectancy of the spacecraft, additional analysis might be warranted in this area to determine the number and complexity of program modifications required to give a sun-only solution for a linear dynamic attitude state model.

In addition, the evaluation of the utility of the magnetometer data will continue to be studied. Daily attitude, definitive attitude, and spacecraft commands will also be issued as required.

CONCLUSIONS AND RECOMMENDATIONS

Magnetometer data currently contains unexplained fluctuations which cause the attitude solution based on magnetometer-only data to vary over a range of approximately 30 degrees in both alpha and delta. Occasionally, and apparently randomly, this solution lies close to the sun cone so that the magnetometer-only and magnetometer-plus-sun solutions are in close agreement. Such a solution is still not necessarily definitive, however, since at other times the magnetometer-only solution again lies close to the sun cone, but an appreciably different sector of alpha, delta space. Here also the magnetometer-only and magnetometer-plus-sun solution agree, yet it is not possible to explain the change from the preceding point of agreement from dynamic considerations. It is concluded that any solution which includes magnetometer data under these conditions should be regarded as only tentative.

No roll data is currently being received, which further strengthens the dependence of attitude determination on magnetometer and sun data. If the magnetometer data continues to be erratic as previously described, the attitude solution may ultimately depend on sun data alone. It is therefore recommended that additional analysis be undertaken to determine the number and complexity of program modifications required to permit a sun-only attitude solution, assuming a linear dynamic attitude state model.

If attitude becomes ultimately dependent on sun data alone, it does not appear possible to continue to satisfy the "mean orbit normal" constraint for nominal attitude. To do so under current geometry would result in loss of sun data and possibly of attitude--assuming continued noisy magnetometer data. It is therefore recommended that for the present the orbit normal constraint be sufficiently relaxed to permit continued reception of sun data.

Task 7

NIMBUS

DISCUSSION

Operational support for the NIMBUS II satellite continued without incident. Two special requests were made for this quarter, a NIMSUM covering the first six days of this satellite and a special definitive WMSAD from 15 May to 1 September.

PROGRAM FOR NEXT REPORTING INTERVAL

Operational support will continue as scheduled and any special requests will be handled promptly and efficiently.

Task 8

OSO-E1

DISCUSSION

The OSO-E1 spacecraft was launched into orbit on 8 March 1967 at 1612 UT. A nominal orbit was achieved with nominal injection attitude. Good data was being returned from the spacecraft shortly after launch.

Prior to launch, an extensive review of the attitude determination algorithm was performed. This was necessitated by the fact that a test tape received from Ball Brothers Research Corporation (BBRC), the spacecraft contractor, was producing attitude results which were not in agreement with the reported test attitude. To provide a cross-check, an alternate method of computing the spin rate was proposed and included in the system. In addition, another method of computing attitude was proposed, and an effort is currently under way to have it implemented as a backup system.

Other work this quarter went into the preparation of a prelaunch analysis, delivered on 28 February 1967 and an early orbit report delivered on 10 March 1967 (see Task 2). Conversion to the System/360 computer was begun in order to phase out the 7094 now in use.

Launch

The OSO-E1 spacecraft was launched from Cape Kennedy on 8 March 1967 at 1612 UT, as scheduled. The orbit and attitude were both nominal and the spacecraft was transmitting good data from the initial orbit.

Initial attitude from orbits 1 and 12 was determined as being near nominal, i.e., right ascension, α , of 78 degrees, and declination, δ of -28°. Later attitude calculations showed an apparently unexplainable change in the declination of some 6 degrees, with no apparent change in right ascension. Reviewing the decommutation part of the system showed that an error condition could arise in the process of time tagging the data. The correlation error would likely to occur when time is imput to the system and not decoded from the real-time data. When time is input to the programs, an associated frame counter value must also be input. (A frame counter is a data word occurring every 25.6 seconds, or every 40 main frames of data.) Its value has little analytic significance except that it is increased by one every time it is stored. Being strictly a function of time it is used for time-tagging the data. After the playback data is decommutated, a search is initiated through the frame counters in the data for the input frame counter value. When found within a given input tolerance, a data time correlation is performed. As it happened, the frame counter input was not the last one found in the playback data. The time associated with the data was in error by some integer multiple of 25.6 seconds.

The effect this has on attitude is limited to its effect on the value of the theta count. The theta count represents a measure of the angle between the sun vector and the projection of the magnetic field vector in the wheel plane. (See Figure 8.) Since the spin rate and pitch angle are nearly constant throughout any orbit (barring a spin-up or pitch maneuver) a time error in their values of 51.2 seconds would be undetectable. Likewise, the SORE count value depends on an equal time difference from one to the other. The SORE count is a count of the clock pulses occurring between the beginning of a certain telemetry word and the time the solar sensor detects the sun pulse. Since the theta (θ) angle depends on the field vector and the field vector on satellite position, a time difference of 51.2 seconds could change the position in the orbit so that a noticeable change in the angle θ would be realized. The theta angle is the most sensitive unknown in the calculation of the roll angle, ϕ, so a change in θ could affect ϕ noticeably. In the particular geometry situation of 21 March, when the sun's declination is near zero, a change in roll will cause a change in the geocentric declination, δ. This was the situation experienced on the data from some of the early orbits.

After the time error was corrected, the determined attitude proved to be nearly nominal $\alpha = 80^{o}$, $\delta = -28^{o}$. Residual biases were chosen such that

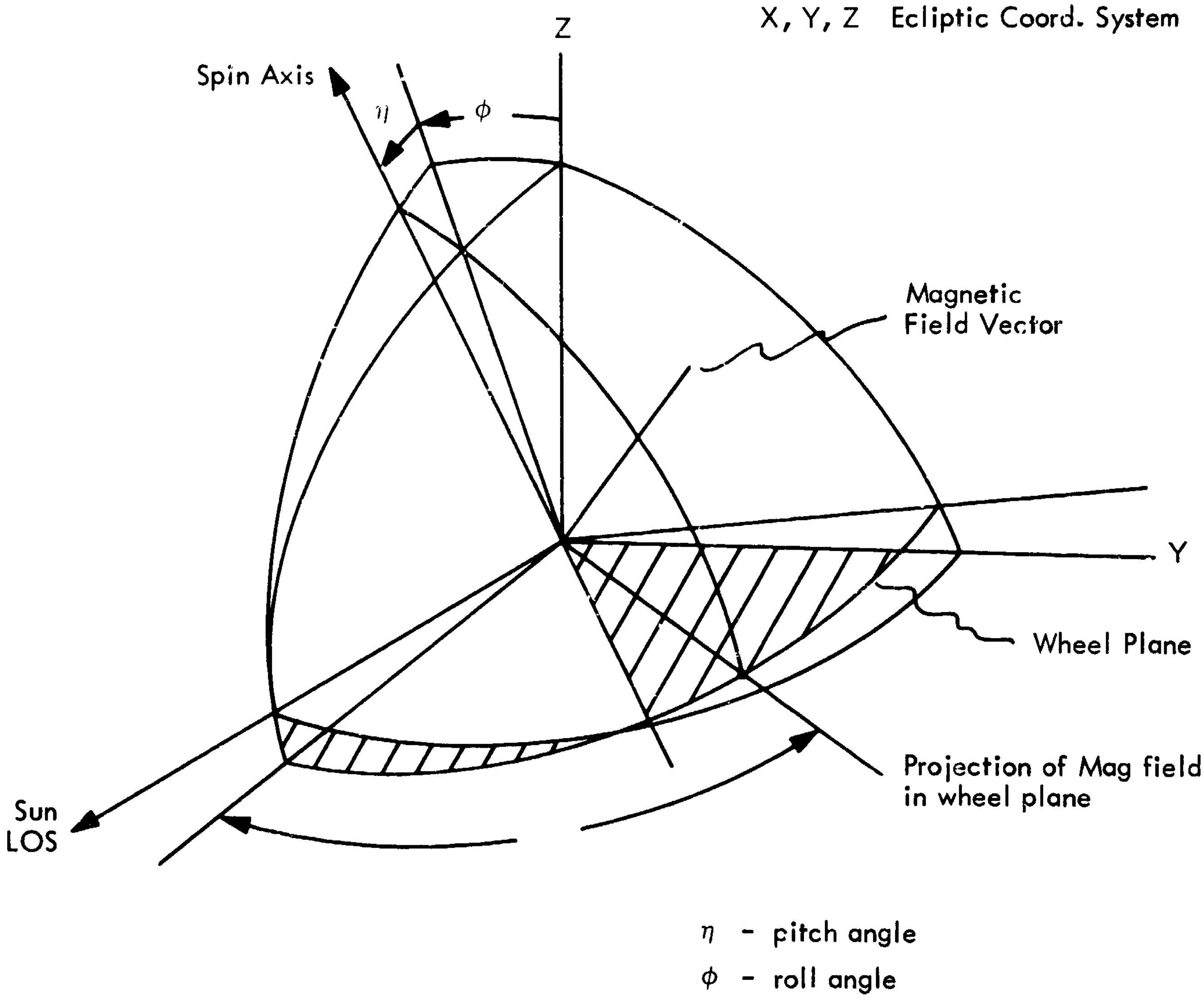


Figure 8. X, Y, Z Ecliptic Coordinate System for OSO-E1

the Magnetic Attitude Predictor (MGAP) model closely followed the determined attitude before and after pitch maneuvers.

Prelaunch

A tape was received from BBRC, which contained telemetry test data obtained during simulation exercises performed in April 1966 at Boulder, Colorado. Along with the data, a spin axis orientation, and all other necessary information needed to process the data, was received.

Several attempts were made to determine an attitude from the data and each time the determined attitude differed markedly from the given attitude. A thorough check was given to all parts of the system and every program calculation was made by hand, and the same attitude resulted. The reason for the discrepancy was narrowed down to the spin rate obtained from the data and the actual spin rate during testing. It was found that the spin rate in the data was monitored and not simulated. Using the modified spin rate from BBRC the given attitude was determined. This check on the system showed where some problem areas could be expected. Thus, an alternate method of computing spin rate (by SORE count) was incorporated, as described below.

The SORE count is a count of the number of clock pulses occurring between the occurrence of a given telemetry word and the sensing of the next sun pulse. Essentially it is a measure of time (see Figure 9).

The counter used to measure the elapsed time cycles at 400 cps. Thus, the time measured difference is derived from the count by the simple formula:

$$\Delta t_i = \frac{C_i}{400}$$

where C_i is the SORE count reading.

Let T be the period of one spin, and N be the number of wheel revolutions between N_o and N_f. The time elapsed during these N revolutions is obviously,

$$25.6 \text{ seconds} - \Delta t_1 + \Delta t_2 \tag{27}$$

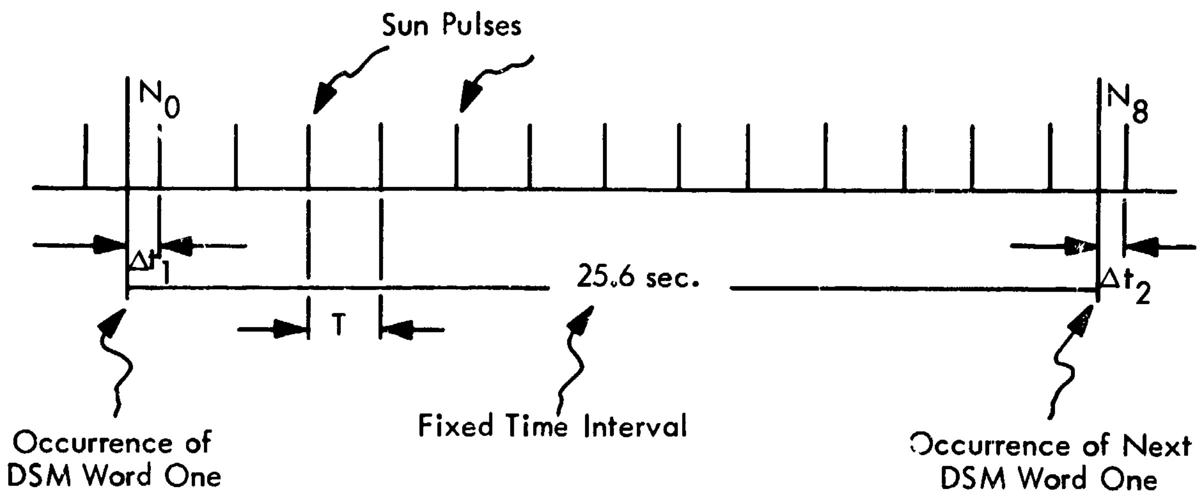


Figure 9. SORE Count Measurement

so that T becomes

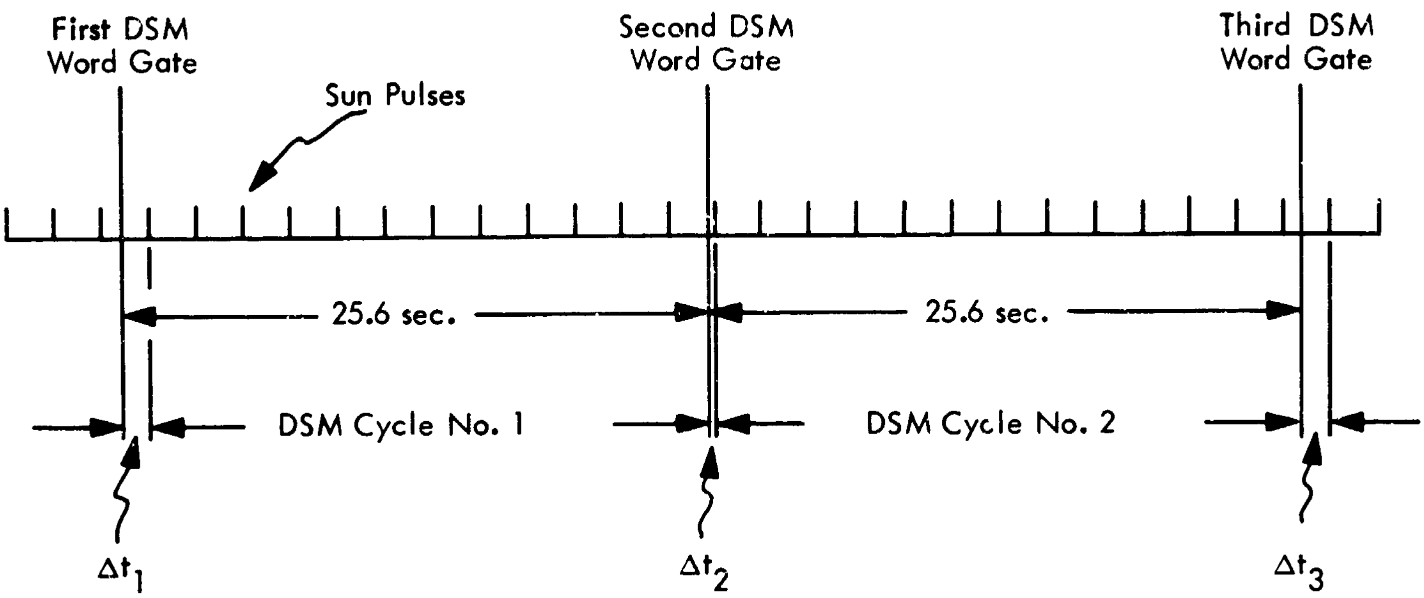
$$T = \frac{25.6 - \Delta t_1 + \Delta t_2}{N} \text{ seconds} \tag{28}$$

Five cases will now be considered regarding the relative magnitudes of three successive readings of the SORE count. Since the SORE count, C_i, is directly proportional to the time difference, Δt_i, Δt_i will be used instead of C_i.

<u>Case 1</u>

$$\Delta t_1 > \Delta t_2 > \Delta t_3$$

When this case arises, it is easily seen (Figure 10) that the number of revolutions between the first and second word gates (occurrence of DSM WORD ONE for a particular cycle) is one less than the number between the second and third.



*Word gate simply means occurrence of DSM WORD ONE for a particular cycle.

Figure 10. SORE Count Measurement, Case 1

From Equation 28

$$T = \frac{25.6 - \Delta t_1 + \Delta t_2}{N} \tag{29}$$

and

$$T = \frac{25.6 - \Delta t_2 + \Delta t_3}{N + 1} \tag{30}$$

or

$$N*T = 25.6 - \Delta t_1 + \Delta t_2 \tag{31}$$

$$N*T + T = 25.6 - \Delta t_2 + \Delta t_3 \tag{32}$$

So that, Equation 32 minus Equation 31 yields

$$T = \Delta t_3 - 2\Delta t_2 + \Delta t_1 \tag{33}$$

Case 2

$\Delta t_1 < \Delta t_2 < \Delta t_3$

In this case (Figure 11), the number of revolutions in the first DSM cycle is one more than the number occurring in the second cycle. So that, Equation 28 gives

$$T = \frac{25.6 - \Delta t_1 + \Delta t_2}{N} \tag{34}$$

and

$$T = \frac{25.6 - \Delta t_2 + \Delta t_3}{N - 1} \tag{35}$$

or,

$$N * T = 25.6 - \Delta t_1 + \Delta t_2 \tag{36}$$

$$N * T - T = 25.6 - \Delta t_2 + \Delta t_3 \tag{37}$$

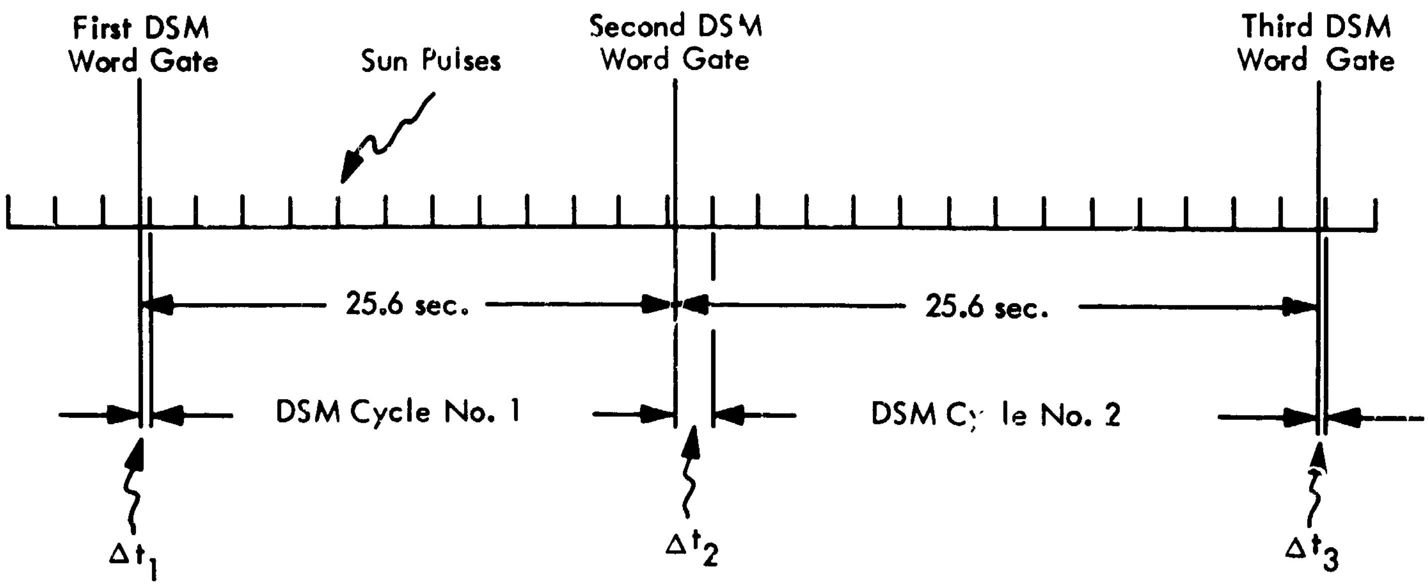


Figure 11. SORE Count Measurement, Case 2

so that, Equation 36 minus Equation 37 yields

$$T = -\Delta t_3 + 2\Delta t_2 - \Delta t_1 \tag{38}$$

Summarizing Cases 1 and 2, Case 2 merely involves a sign change from Case 1. Thus, we have

$$T = ABS\ (2\Delta t_2 - (\Delta t_3 + \Delta t_1)), \tag{39}$$

as a general formula for calculation of the spin period when either Case 1 or Case 2 occurs.

The final three cases are:

CASE 3 $t_1 < t_2 < t_3$

CASE 4 $t_1 = t_2 = t_3$

CASE 5 $t_1 > t_2 > t_3$

Figure 12 shows that in any of these cases, the number of revolutions between the sun pulses used to measure the successive SORE counts is constant.

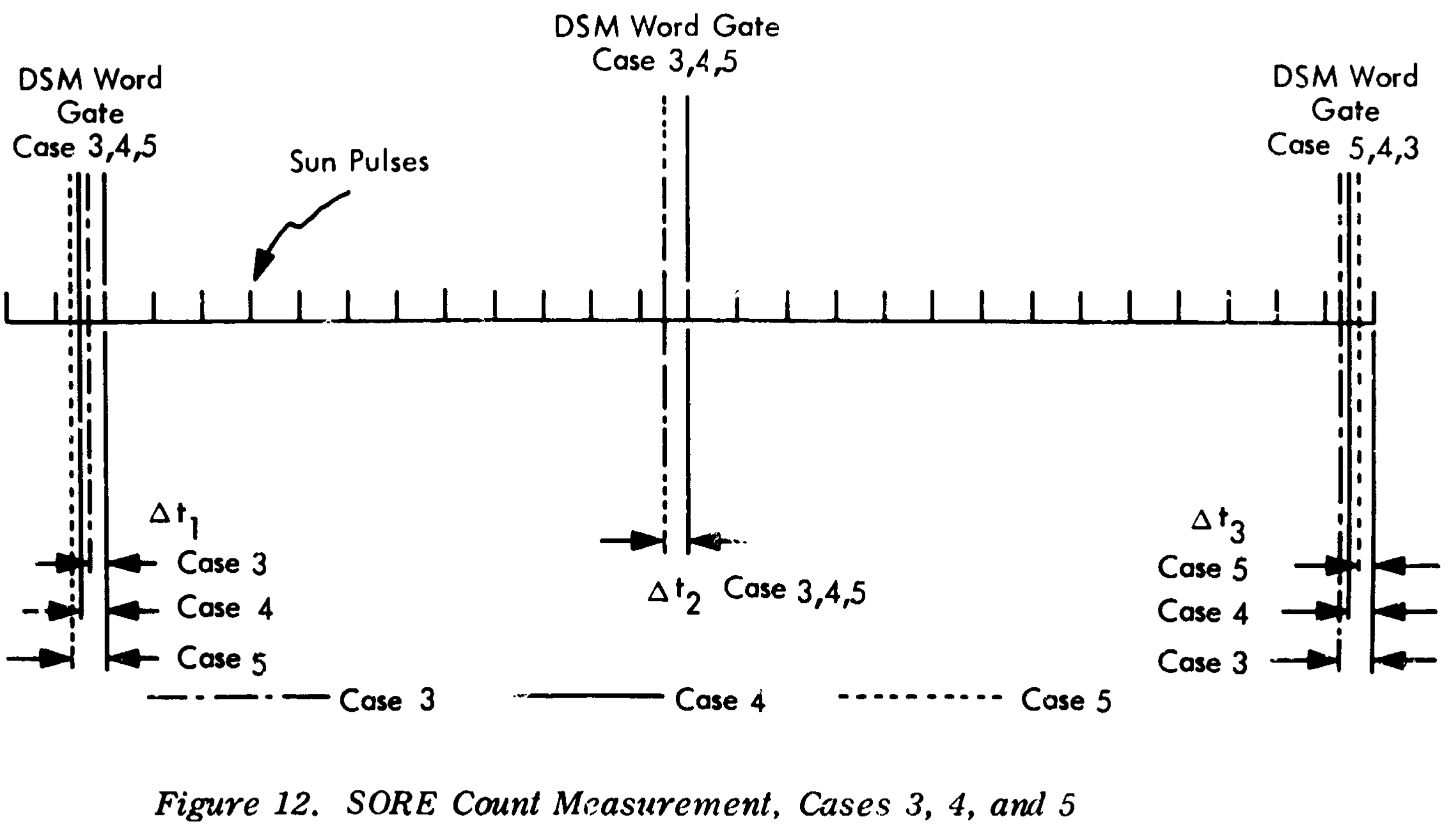


Figure 12. SORE Count Measurement, Cases 3, 4, and 5

Eliminating N from two successive equations will also eliminate T. T is thus indeterminant in any of these cases.

The range of the spin rate is from 2.69 sec/rev. to 1.51 sec/rev. In a 25.6 second interval the wheel could spin, under nominal conditions, from 9.5 times to 17.0 times. Thus, the number of revolutions is near one of the integers, 10, 11, . . . , 17 the spin period would be indeterminant for the reasons previously indicated. This system of determining the spin rate can be used as backup if the current method is questioned.

System/360 Conversion

Three subroutines have been written for the System/360 conversion effort. One routine handles the reading of the raw input tape and is the only machine language program left in the system. A large machine language program, RUNT, has been coded in FORTRAN. RUNT performed the data unpacking in the decommutation segment of the system. With the completion of these two routines, system testing of the decommutation segment was started.

A routine has also been written to reformat data output from the 7094 decommutation program (DECOM) to an acceptable format for the System/360. This permitted system testing of the Attitude Determination algorithm (OSOAP) prior to a satisfactory decommutation system. These two system segments and the magnetic attitude predictor (MGAP) are in system testing. An operational 360 system is planned.

Alternate Attitude Determination Method with no Pitch Values

An alternate method for determining the attitude of the OSO-E satellite was studied. This attitude scheme is a differential correction process using the observed theta angle to correct a given attitude. It differs from the attitude program, OSOAP, because it requires only the theta angle, and not the pitch angle observations. This attitude method has applications as both an attitude determination system and an analysis tool to determine possible satellite generated magnetic fields.

The previously mentioned attitude scheme is as follows. An initial spin vector, $\vec{S}_0$, and the observed theta angles, θ_{i_0}, $i = 1, \ldots n$ are obtained from the OSO DECOM program. The theta angle is the angle between the vector to the sun and the projection of the geomagnetic field vector onto the wheel plane of the satellite measured counterclockwise from the vector to the sun about the spin axis. Then mathematically the $\cos \theta_{ic}$ are computed using the equation

$$F_i = (\vec{S}_i \times \vec{H}_i)^* \cdot (\vec{S}_i \times \vec{G}_i)^* = \cos \theta_{ic}$$

where

$\vec{S}$ is the spin vector of the satellite

$\vec{H}$ is the geomagnetic field vector

$\vec{G}$ is the vector to the sun and the asterisk signifies that these cross product vectors are normalized.

Define Q_i as

$$Q_i = \left(\frac{\partial F}{\partial X}\right)_i \Delta X + \left(\frac{\partial F}{\partial Y}\right)_i \Delta Y + \left(\frac{\partial F}{\partial Z}\right)_i \Delta Z = \cos \theta_{io} - \cos \theta_{ic} .$$

Using the least square method to solve for ΔX, ΔY, ΔZ represented by ΔV, gives the matrix notation $\Delta V = (\partial F^T \partial F)^{-1} \partial F^T Q$. Defining $\vec{S}_i$ as $\vec{S}_i = \vec{S}_{i-1} + \Delta \vec{V}$, this iterative differential correction of the spin vector is continued until $\cos^{-1} (\vec{S}_i^* \cdot \vec{S}_{i-1}^*) < \Delta c$, where Δc is the convergence criterion. A maximum number of M iterations is allowed in case this process will not converge due to a bad guess of the initial spin vector. This attitude scheme has been programmed and is currently being tested on the S/360 computer system.

PROGRAM FOR NEXT REPORTING INTERVAL

The completion of the system conversion to System/360 is planned for the early part of the next period.

Any necessary system changes required to support the OSO-D spacecraft, to be launched later this year, will be made. A study will be made to determine the optimum launch data for the OSO-D spacecraft. The optimum time is such that the Ft. Myers ground station will never have to acquire both spacecrafts (OSO-E1 and OSO-D) at the same time.

The documentation of the system is planned for completion and documentation changes will be made to reflect the conversion to System/360.

Task 9

TOS/ESSA

DISCUSSION

Operational Support

On 23 January 1967 a request was received from NASA to add the ROSMAN CDA station to all operational program runs. This request was complied with. In the near future the ROSMAN station is expected to take over the TIROS duties from the WALOMS CDA station and WALOMS will remain as a standby station only.

The TOS-B (ESSA-4) satellite was launched on 26 January 1967. The TOS/ESSA Attitude Determination System supplied excellent attitude results from the initial spacecraft acquisition on orbit 1 until control of the satellite was passed to ESSA on 8 February 1967. One small discrepancy was noted in the TOS/ESSA ADS system output due to an error in the TOS-B Calibration and Alignment Data Handbook. Four telemetry channels (13, 14, 15 and 16) did not have a minus sign printed in front of the voltage readings. This discrepancy was corrected on orbit 1 and no further anomalies were noted in the processing system. TEC/TTCC was very impressed by the improved accuracy of the new floating point relocation and calibration routines in STADEE, the housekeeping telemetry evaluation program.

One spacecraft malfunction was detected soon after the launch of TOS-B. The gamma (sun angle) readings from the Digital Solar Aspect Indicator on the satellite and the gamma angles computed from the horizon sensor attitude results did not always agree. An analysis of the available data was conducted to determine the cause of the error. It was determined that binary bit 2 (next to the most significant bit) in the DSAI word was always reading out a zero. If an angle was sensed that did not use this bit, the reading was accurate. However, when an angle was read out that used bit 2, the reading was in error by approximately plus or minus two degrees.

Daily attitude results, predictive MGAP steering, and housekeeping telemetry data from TOS-B were supplied to TEC/TTCC until ESSA assumed the responsibility for the spacecraft on 8 February 1967. Also, a 180-day gamma angle study, using actual post-launch elements for TOS-B, was supplied to Mr. R. D. Werking (NASA), as requested.

Attitude determination was provided on the TIROS VII, VIII, IX, X, and ESSA-4 spacecrafts. The Theory and Analysis Office reassumed processing of the TIROS IX satellite data on 8 February 1967, as TEC/TTCC reassumed control of the satellite on this date. Scheduling of contacts with the TIROS VIII spacecraft has been altered to the following pattern: two weeks with no contacts, and 48 hours of tracking for orbital and attitude data; the cycle is then repeated.

A total of 629 messages were received and processed through the ADS with the following results:

1. TIROS VII, 285 messages yielded 187 attitude results.
2. TIROS VIII, 45 messages yielded 9 attitude results.
3. TIROS IX, 59 messages yielded 50 attitude results.
4. TIROS X, 175 messages yielded 140 attitude results.
5. ESSA-4, 65 messages yielded 56 attitude results.

The messages that did not yield spin axis solutions were unusable due to noise, low or no amplitude horizon crossings, incorrect time codes, format errors or physically bad tapes from TEC/TTCC.

Operational problems were not encountered with any of the satellites except TIROS VII. Attitude reduction and prediction continue to be a serious problem with this spacecraft. Sun interference precluded attitude determination for some periods, and indiscernible horizon crossings during extreme NON conditions (very high or low) seriously hampered attitude processing in other periods. At one point, the horizon crossings were unusable for approximately ten days due to insufficient amplitudes. During this time, TEC/TTC was contacted to supply hand-reduced scan ratios (earth scan length divided by spin period) to use in the ROLLS program. The solutions from these hand-reduced ratios verified predicted spin axis movement through this interval.

The spin rate on this satellite has decayed to 2.3 rpm, and MGAP predictions for periods of time exceeding one week are virtually impossible. Consequently, this creates a scheduling problem because orbits needed for attitude data must be scheduled at least two weeks in advance. If the orbits are to be scheduled on erroneous predictions (which cannot be helped), then they may or may not contain usable attitude data. When all of these contingencies are compounded, attitude determination and prediction require constant monitoring of the data to preclude the loss of the spacecraft. Every effort will be made to extend the life of TIROS VII as long as possible.

Programming and Analysis

A discrepancy in the confidence level computations in the ALS (Attitude Least Squares) program was noted and corrected. This apparently was a long standing error that was recently discovered. Also, a new ROLLS program was written and debugged. This program computes roll angles using any combination of eight different methods. This program was written to replace the obsolete program SRAC (Scan to Roll Angle Conversion). A program DATGEN (Data Generator) was also written to generate input data to test the ROLLS program. ROLLS is now being used on an operational basis to process hand-reduced scan ratios from TEC/TTCC.

The operational "four-wire" DATPRO was modified to allow processing of any value spin period for a TIROS/ESSA series satellite. This new version precludes the need for constantly modifying the program to handle the exceedingly large spin period of TIROS VII. A subroutine CORECT was also written for eventual use in DATPRO. CORECT adjusts the earth scan lengths in DATPRO to compensate for orbital angular velocity. This subroutine is currently undergoing testing and analysis with an analogous subroutine, DELPHI, which is used in ALS.

An investigation of a buffer-sharing process in DATPRO has been initiated. If this process is successful, several hundred core locations could be saved. This process would be particularly desirable in the "five-wire" DATPRO, as the sun-masking routines barely fit into core. This technique was required because the ESSA satellites have an IR horizon sensor that is mounted very close to the operationally desired gamma angle. Consequently, this sensor is "seeing" the sun, as well as the earth, a large percentage of times when the satellite is in sunlight. The subroutine SUNB5 was added to DATPRO. SUNB5 computes from the a priori attitude the expected earth scan length, and "sun to earth angle" if sun interference is expected. An additional advantage to this technique is that it inherently avoids interference from the orthogonal sensor present on some TOS satellites.

The program RENUMB (re-numbering program) has been completed and the documentation sent for publication. This program renumbers the statement numbers in a FORTRAN II or IV program into an ascending numerical sequence to facilitate following the logical flow of the program.

A new method was added to STADEE (Status Data Extraction and Evaluation Program) of the five-wire ADS to increase the accuracy of the telemetry. This modification eliminated errors introduced by noise, preventing the telemetry points to remain at a constant level. New formulas were also added to compute the calibration and relocation factors. Prior to launch, test runs were made with ESSA-2 and ESSA-3 data. These results compared very favorably with the Sanborn recordings at TEC/TTC.

ESSA IV

DTO (Detailed Test Objectives) and the launch date (25 January 1967) were received from NASA on 5 January 1967, to begin the prelaunch analysis. All necessary computer runs and analyses for the prelaunch report were made. The first rough draft of the prelaunch analysis was completed 13 January 1967 and the complete printed document was distributed (by NASA) on 25 January 1967 (see Task 2).

As mentioned perviously, the five-wire ADS was utilized successfully in support of ESSA-4 from the first orbit on 26 January 1967 until control was given to ESSA on 8 February 1967. This was the first time the five-wire ADS was used operationally, and no problems were encountered in either attitude determination or acquisition of the housekeeping telemetry.

TOS-C

A prelaunch analysis for TOS-C (scheduled for launch on 19 April) is under way and will culminate in a report published prior to the launch. The first item of this analysis is a conflicts study, similar to one performed between ESSA-2 and TOS-B. The conflicts study was requested for ESSA-3 and TOS-C and was completed 15 March 1967. This study showed the location of ESSA-3 with respect to TOS-C from 12 April to 12 May 1967, assuming TOS-C is launched such that the ascending node will occur at 1500 local time. The study also showed days TOS-C could be launched to fulfill the constraint that ESSA-3 and TOS-C should be on opposite sides of the earth. If this constraint is fulfilled, maximum picture coverage from these two AVCS satellites will be achieved.

The five-wire ADS system is being prepared for the TOS-C launch Program. STADEE has been modified so that it will process up to and including ESSA-10, provided all satellites are launched successfully in order. The programs DATPRO and ALS-ARS needed no updating since these programs already had the capacity to process TOS-C data. Initial test runs have begun and system tapes are being generated.

PROGRAM FOR NEXT REPORTING INTERVAL

Attitude determination, attitude control, attitude prediction and operational support will be continued for the TIROS VII, VIII, IX, X, and ESSA-5 (TOS-C) satellites. Special attention will continue on TIROS VII to extend the lifetime of the spacecraft as long as possible. Full support will be supplied for the TOS-C launch scheduled for 19 April 1967, and any other launches that may occur. The programming systems will continue to be improved as needed in this area.

Task 10/23

GREMEX

DISCUSSION

All routines for the GREMEX (Goddard Research Management Exercise) were unit tested. The flow charting and documentation drafts were completed. Test runs were made with the GREMEX 7094 system for purposes of comparison with the GREMEX System/360 programs. Tests were run on the 360 program system and the comparisons made.

The task was completed with delivery of the following items to Mr. M. Denault, Manager, Management Information Systems Branch, on 2 February 1967.

During March 1967, a call was received from Mr. Denault requesting help in making some changes in the GREMEX 7094 system and re-organizing the drafts of the 360 system. A one-day workshop was set up to help familiarize the NASA/GREMEX operations personnel with the system. Another call has been received concerning possible problems the GREMEX System/360 program. One-day evaluation period was arranged for determining the extent of the problems. The results were not available at the time of the preparation of this report.

PROGRAM FOR NEXT REPORTING INTERVAL

The task was considered complete on 2 February 1967 with the delivery of the material listed above. However, with the requests for help in using the program, it is expected that the task may be amended to help the NASA personnel responsible for use of the program.

CONCLUSIONS AND RECOMMENDATIONS

The GREMEX system is a fairly large complex program which probably has not been thoroughly tested through all plays. The plan appeared to be that

NASA/Goddard would take over the program and be prepared to run it. However, to do so requires at least one experienced programmer with a thorough knowledge of the program. Preferably for insurance there should be several such persons available to obtain this knowledge, which takes time and much exercising of the program. There do not appear to be such persons available. Therefore, it is recommended that several persons be trained to use and make changes to the system as quickly as possible.

Task 11/20

DEFINITIVE ORBIT OPERATIONS AND ANALYSIS

DISCUSSION

Efforts were concentrated on the processing of data from the three satellites reported as of 1 January 1967 (QPR-2). Although the priority of work varied throughout the period, completion of definitive orbital results for these three satellites, POGO, GEOS, and ATS-1, was of primary concern. Figure 13 summarizes the production results for these satellites indicating the status as of 1 January 1967 and the major achievements during the quarter.

Of special importance to the successful completion of this task has been the efficient utilization of available 7094 computer time (K computer), and the scheduling required to maintain continuous operation on this machine, 24 hours per day. Daily monitoring of the job priorities, and subsequent scheduling of all phases of the work load has been completed by the group leader. In addition, special attention has been given to the preparation of weekend computer utilization both in terms of manpower scheduling and sufficient workload for the computer. Hence, overtime manpower has been scheduled every weekend to ensure that satisfactory performance and results are achieved. The following paragraphs describe the progress and results achieved by satellite during the quarter.

POGO (OGO-II

The primary objectives of this quarter were completing all phases of processing from 21 November 1965 to 1 March 1966 by 1 February 1967, and although some difficulties were encountered, this result was achieved as desired. This total job consisted of converging 100 differential correction (DC) runs (two-day arcs with one-day overlap), completion of the error analysis and

CONCLUSIONS AND RECOMMENDATIONS

The TIROS VII spacecraft continues to require constant attention to preclude the loss of the satellite. A major effort will continue in this area in order to cope with the anomalies in the data and scheduling problems.

TIROS VIII is now on a semi-monthly contact schedule, and for all practical purposes is a "dead" satellite. However, until the Theory and Analysis Office directs otherwise, operational support and attitude determination will continue as the data becomes available.

The five-wire hardware at TEC/TTCC should undergo thorough tests again before the launch of TOS-C to ensure proper operation of the equipment in support of the launch.

record keeping necessary to ensure satisfactory results and final generation of an orbital vector position tape for the experimenter.

The required error analysis consisted of two steps: 1, orbit position vector comparison between successive arcs and the recording of results necessary to generate a final output tape (Figure 14) and 2, analysis and recording of range/ range rate data, by station pass, from each DC to ensure satisfactory fit of this data and that data sets were compatible between arcs (Figure 15). Figures 14, 15, 16 and 17 are samples of various charts used for record keeping developed for this and other satellites under consideration.

Upon completion of the data for the period outlined above (21 November 1965 to 1 March 1966), POGO work was terminated and did not receive priority again until March. Since that time, 172 additional arcs (3 March to 24 August 1966) have been converged in the DC and error analysis completed. In all, 272 days of data were processed during the quarter.

Special effort was also given a request to complete differential corrections and error analysis on the first nine days of data for this satellite. This occurred during the period of satellite degaussing, and therefore required short runs between these maneuvers. This job was completed as requested and necessitated overtime manpower and weekend services.

GEOS

Except for a two-week period in February, the objective for GEOS was completion of DC runs on a low priority basis serving to absorb unused computer time. These arcs were the standard two-day arcs with one-day overlap, and using all observation types (Minitrack, optical and range).

In February, priority centered on this satellite and 120 arcs were successfully converged. Analysis of the first 200 arcs completed indicated many cases with excessive observation data being rejected by the DC. As a result, a procedure was devised to rerun the poorest cases (60 arcs) and special effort expended on a weekend to rerun and re-record these arcs.

At the end of the quarter, 280 differential corrections (period ending 1 August 1966) were completed (using all data types) and the error analysis procedures completed through 210 (6/3/66).

ATS-1

Responsibility for the generation of final definitive results for this satellite was assumed in mid-January, and work begun immediately. The project objective required rerunning all data since launch (7 December 1966), to bring results up-to-date as rapidly as possible then, remain current on a weekly basis. Since the first two weeks of satellite life consisted of many orbital maneuvers, the procedure consisted of generating short arcs between maneuvers. After this period, the arcs were eight days in length with one day overlap. These eight-day arcs were generated and updated on a daily basis and orbital results delivered on the ninth day. Considerable overtime effort and manpower was expended to complete this task in an expedient manner. By 27 January all work was current, and data has been delivered on schedule since that time.

Many special computer runs and analysis tasks have been completed throughout the quarter in an effort to supply the project the best possible definitive data.

PROGRAM FOR THE NEXT REPORTING PERIOD

a. The ATS-1 definitive orbit production and analysis effort will be continued on a top priority basis; and the current orbital data will be supplied weekly to the project.

b. When the ATS-A is launched responsibility will be assumed for definitive orbit results on a current weekly basis.

c. The first phase of processing POGO data will be completed through December 1966.

d. The research and analysis task of determining station biases and station positions using differential correction on GEOS observation data will begin. Also GEOS data will continue to be processed on a low priority basis.

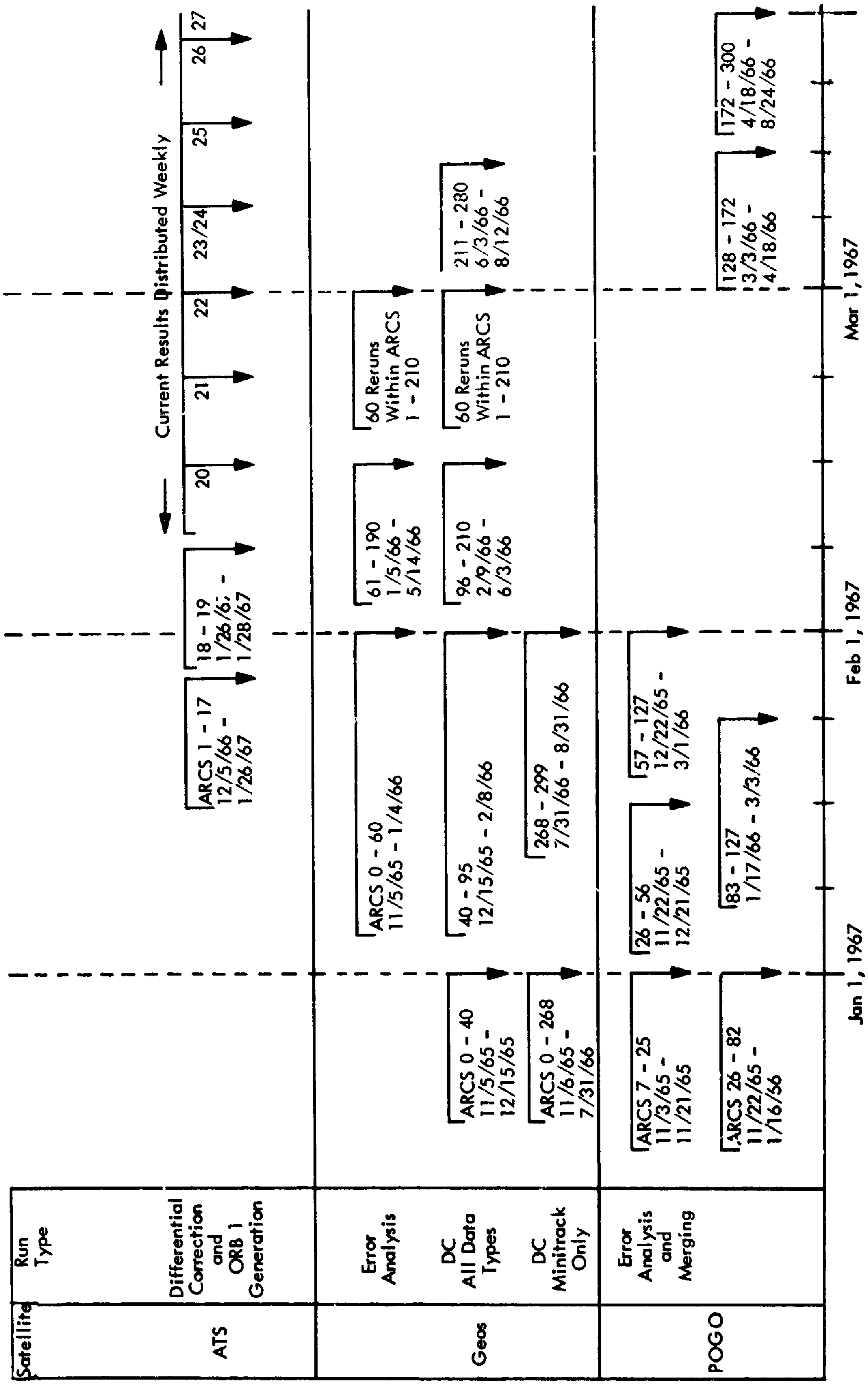


Figure 13. Definitive Orbit Differential Correction Operations Summary 1st Quarter 1967

VECTOR DIFFERENCE - ERROR SUMMARY

	Common Period																		
ARC #	Start Date	Hour	Stop Date	Hour		Minimum Delta Range		Time of Minimum			Maximum Delta Range		Time of Maximum		Date Run				

Figure 14. Definitive Orbit - Ephemeris Comparison Report

RANGE DATA STATION ERROR ANALYSIS SUMMARY

Date and Station Data						Range Data						Range Rate Data							
	Pass Start	Pass Stop				Case A			Case B			Case A			Case B				
Date YrMoDa	HrMin	HrMin		Station Name		Total OBS	No. in Solution	AVG WT'D Rms	No. in Solution	WT'D Rms		Total OBS	No. in Solution	AVG WT'D Rms	No. in Solution	WT'D Rms			

Figure 15. Definitive Orbit Range/Range Rate Overlap Comparison Summary

STATUS REPORT

D. C.							ERROR ANALYSIS						TAPES						
ARC #	Date Submitted	Date Completed	Sigma	Sigma	Conditions of Run		RHO	ORB-1 A	ORB-2 B	Difference Program			ARC #	D.C. C-1	D.C. B-1	ORB-1 D.P. A	ORB-1 D.P. B	ORB-1 C-2	Diff. A-5

Figure 16. Definitive Orbit - Differential Correction Status Summary Report

DEFINITIVE ORBIT OPERATIONS REPORT

NAME:	ARC#	DATE:	EPOCH

PERIOD FROM: TO:

INPUT PARAMETERS

RHO:	SIGMA:	Δ SIGMA:
X_1:	Y_1:	Z_1:
X_F:	Y_F:	Z_F:
$XDOT_1$:	$YDOT_1$:	$ZDOT_1$:
$XDOT_F$:	$YDOT_F$:	$ZDOT_F$:

SAT. REFERENCE:

OBSERVATIONS

TYPE	TOTAL NO.	ACCEPTED	% ACCEPTED	R. M. S.	
					COMBINED
					TOTAL ________
					TOTAL NO.
					ACCEPTED ____

D. C. PROGRAM

INPUT	SORTED OBSER.	OUTPUT	UPDATED ELE.
	CON. EDIT OBSER.		BCD OUTPUT TAPE

COMMENTS:

APPROVAL: ____________________

Figure 17. Definitive Orbit - Differential Correction Arc Summary

Task 13

LAUNCH WINDOW

DISCUSSION

A launch window study, initiated in February, is being conducted for the IMP F satellite. Two previous launch window studies for IMP F have been completed, but launch delays and changes to injection parameters have necessitated a third.

The initial range of this launch window study was 15 April to 15 July 1967. For every fifth day throughout this range, Launch Window Program runs were made at 0.2 hour increments from two hours preceding to two hours following the injection time resulting in a spin axis ecliptic plane angle of ninety degrees. The results of the runs provided an initial approximation of the one-year lifetime line and all required data on the shadow constraint.

Unfortunately, this initial approximation of the one-year lifetime line created some doubt as to whether it would be possible to obtain a satisfactory launch window with the current set of injection parameters. A table was prepared for Mr. D. Stewart (NASA), based on Launch Window Program runs, showing the maximum launch window obtainable to be only 22 minutes in length when 3-sigma variations in the injection parameters were considered. A 30-minute launch window is desirable. In addition, the spin axis ecliptic plane angles which would be obtained with this launch window bordered on the unsatisfactory.

The launch window study was delayed for several weeks until a decision was reached by the IMP Project Office to continue the launch window study with the same injection parameters. The range of the study, however, was reduced from three months to 15 days—17 May to 31 May—due to the urgency of the study.

Launch Window Program runs have now been made for each day within this new range, and initial approximations of one-year lifetime lines have been made for both nominal and plus three-sigma injection parameters. PERTAPE I runs are now being submitted to define more precisely these one-year lifetime lines.

A partial check of the lifetime results of the Launch Window Program was made by making a comparison run with PERTAPE I for an acceptable injection time on 15 April 1967. The Launch Window Program predicted a lifetime of 444 days while PERTAPE I predicted 446 days. An equivalent PERTAPE II run was also made which was terminated by the operator after 434 days of output due to excessive running time. However, based on the perigee height after 434 days, it is likely that PERTAPE II would have predicted a lifetime in excess of 500 days. The cause of this discrepancy is not presently known and will require further study.

PROGRAM FOR NEXT REPORTING INTERVAL

The launch window study for the 15-day range mentioned above will be completed shortly.

A launch window study for the three-month range initially requested will be completed (if desired).

CONCLUSIONS AND RECOMMENDATIONS

Launch Window Program results indicated that a 30-minute launch window would be possible for plus three-sigma injection parameters only if the minimum acceptable spin axis ecliptic plane angle was reduced from 85 to 84 degrees. This reduction was agreed to by the IMP Project Office when it requested the continuation of the Launch Window Study without changing the injection parameters.

Task 14

ORBIT OPERATIONS

DISCUSSION

The generation of weekly predictions for station tracking and acquisition, the generation of refined orbital data, prelaunch and launch day support continued to be of primary concern during the first quarter of 1967.

The weekly ORB1 ephemeris tapes and world map data were generated for the Minitrack section on a regular scheduled basis. Control and follow-through procedures were maintained on this phase, and quality checks were made on all output to ensure distribution of correct data. This data was generated for 38 scientific satellites throughout the period, with 83 regularly scheduled programs to process each week, plus any special requests approved by the Orbital Determination section. In addition, a special task was completed in March for the Data System Division. As requested, an analysis of the present operational orbit determination and prediction system (including the Minitrack input/output phases) was made and a standardized reporting procedure was devised for use by all departments concerned. Figure 18 illustrates the flow diagrams of the present operational system, keyed to Charts I - IV (Figures 19 through 22) which are to be used by the four reporting areas. All operations will be reported weekly to give management an overall view of the orbital prediction problems.

During March, special emphasis and priority was given to the generation of refined data for the ALOUETTE 1, ALOUETTE 2, DME 1 and the EPE-D satellites. To meet the special requests, rush projects were completed on the weekends of March 4, 11, and 18, requiring extra overtime and manpower scheduling. All data was processed to completion and the tapes and books were distributed after a quality check. At this time, refined data processing, except air density, is now completed and up-to-date on all satellites.

For refined data output, a new control log and status report was designed, printed, and put into operation. This summary should permit closer control of operations (both for data in process as well as that completed) and provide a permanent record of the recipients of the output books and tapes. This summary (Figure 22) consolidates all information, including period of book, date of generation, permanent tape numbers and to whom the output was delivered and is readily available for all to use in a central location.

Three launches were supported for which prelaunch, launch day and post-launch operations were provided. These launches were INTELSAT on 11 January, ESSA 4 on 26 January and OSO-3 on 8 March 1 . In addition, nominal prelaunch data was processed on fifteen other scientific satellites.

ATS-A	INJUN 4	OGO F
BIOS B	INJUN 5	OSO G
D1 A	INTELSAT F3	RAE A
ESRO 1	IMP F	S 3
ESRO 2	OAO A2	UK 3

PROGRAM FOR THE NEXT REPORTING PERIOD

Predictive orbital data, refined orbital data and nominal prelaunch data will continue to be provided in support of (and as requested by) the Orbital Determination section. Prelaunch and launch day support will be provided for approximately five satellites scheduled for launch in the next period.

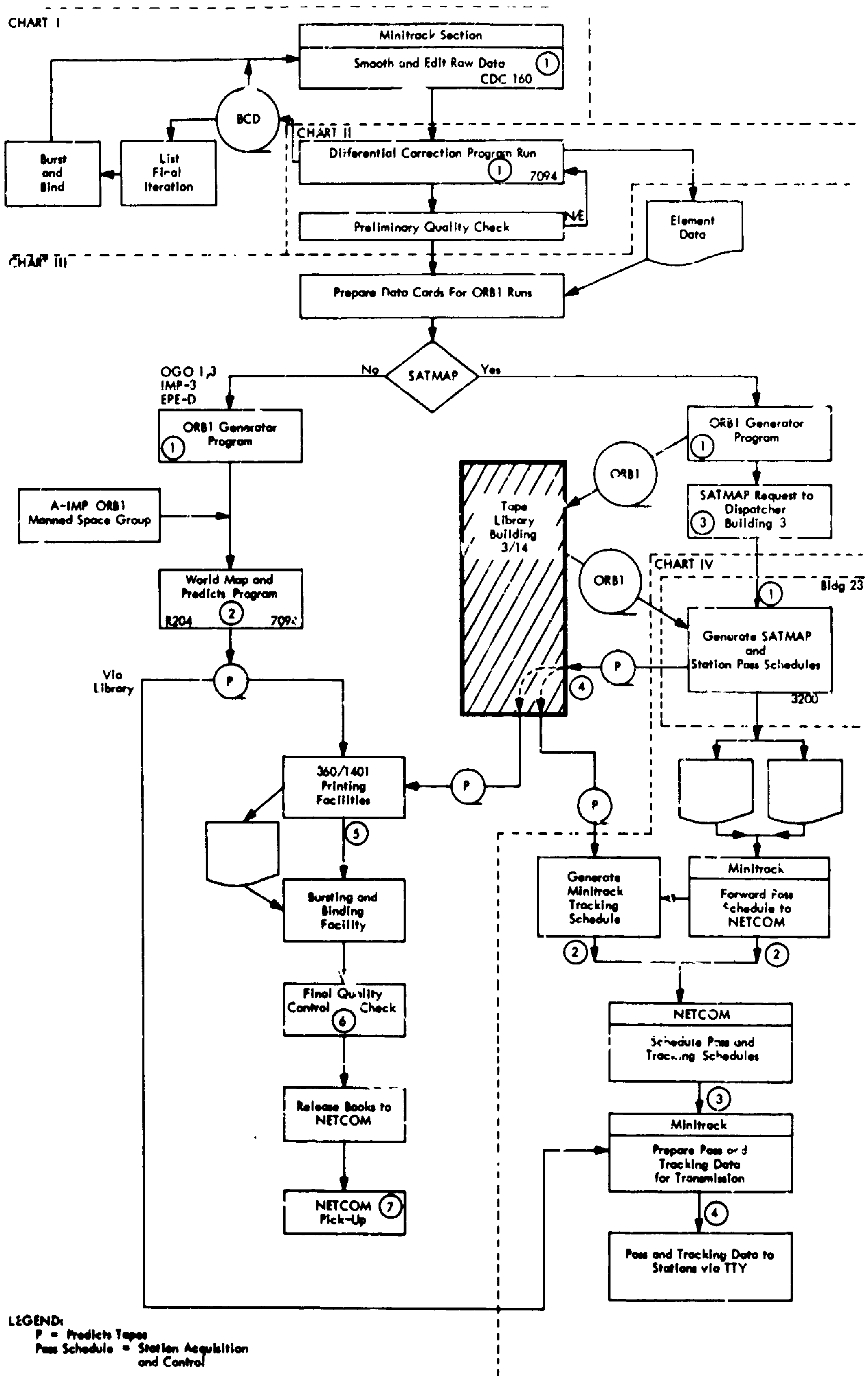


Figure . . GSFC Operational Orbital Determination and Predictive Orbit System

CHART I

MINITRACK INPUT PROCESS

TRACKING WEEK ____________ TO ____________

SATELLITE I.D.	OBSERVATION DATA [1]							[2]	[3]	
	START OBS.	STOP OBS.	TYPE	NUMBER OF PASSES	DAY-HR. DUE	DAY-HR. AVAILABLE		D.C. BOOK AVAILABLE	D.C. TAPE RECEIVED	COMMENTS

Figure 19. Data System Division - Orbital Operations Report - Minitrack Input Section

CHART II **DIFFERENTIAL CORRECTION PROCESS** **TRACKING WEEK__________TO__________**

SATELLITE		ENG. RESP. FOR D.C.	DATE DUE	DAY-HR. RUN	OBSERVATION		OBSERVATION TYPE						PERTAPE		COMMENTS
I.D.	NAME				START	STOP	M-2	M-3	R-1	R-9			N	PERIOD	

Figure 20. Data Systems Division - Orbital Operations Report - Differential Correction Section

CHART III

STATION TRACKING WEEK __________ TO __________

SATELLITE		ORB. I I		WORLD MAP COMPLETE R204 2	DAY-HR. SATMAP REQUEST 3	PREDICTS AVAILABLE DAY-HR. 4	DAY-HR. BOOKS LISTED 5	QUALITY CHECK DATE/BY 6	BOOKS RECEIVED BY NETCOM 7	COMMENTS
I.D.	NAME	DATE DUE	DAY-HR RUN							

Figure 21. Data Systems Division - Orbital Operations Report - Predicts Section

CHART IV

MINITRACK OUTPUT PROCESS

TRACKING WEEK__________ TO__________

SATELLITE I.D.	ORB I AVAILABLE DAY-HR.	1 PREDICTS AND SCHEDULES RUN DAY-HR.	PASS SCHEDULING			TRACKING SCHEDULE		PASSES READY TO TTY DAY-HR. 4	PREDICTS READY TO TTY DAY-HR. 4	COMMENTS
			TO NETCOM DAY-HR. 2	FROM NETCOM DAY-HR 3	ADDED PASSES DAY-HR. 3	TO NETCOM DAY-HR. 2	FROM NETCOM DAY-HR. 3			

Figure 22. *Data Systems Division - Orbital Operations Report - Minitrack Output Section*

REFINED ORBITAL DATA - STATUS SUMMARY

	PERIOD										DELIVERED TAPES			RECEIVED BY		
	Start	Stop	Arc or Book No.	Date Submitted	Pertape No.	ORB-1	ORB-3A	M/ORB	Edit A Tape	M/ORB Listing Rel.	ORB-1	ORB-3A	M/ORB	Date	Name	Date Completed

Figure 23. Orbital Determination - Refined Orbital Prediction Summary

Task 15

SCIENTIFIC DATA ANALYSIS/GENERAL

DISCUSSION

S3, S3A Cosmic Ray Section Data Processing

Figure 24 shows the data flow of the S3, S3A Cosmic Ray Data Processing System (S3DPS). The final draft of the system documentation has been sent out for typing and will be published early in the next quarter.

The Encyclopedia Update (ENCYUD) and Logbook Update (LGBKUD) programs were modified. While processing the S3A Logbook tapes through LGBKUD, it was discovered that the available start times of the data were not precise enough for the definitive orbital data retrieval. The definitive orbital data is retrieved by orbit routines (supplied by the division) for a time specified in the calling program.

Because the available times on the Logbook tapes were not precise enough, LGBKUD could not retrieve the data. Therefore, LGBKUD was modified to read in a summary tape containing the required definitive orbital data. The summary tape was already produced by ENCYUD for the S3 data. ENCYUD was modified to process either S3 or S3A data and produce a summary tape for the LGBKUD program.

While testing ENCYUD, the orbit tape could not be read. Analysis of this tape showed that it had not been deblocked properly from the Direct Couple System (DCS) format. A second deblock attempt was also unsuccessful. The third deblock produced a good orbit tape.

Testing of ENCYUD was resumed using the new orbit tape; however, the orbit routines were not functioning correctly. The problem was brought to the attention of the division. A modified orbit routine was tested, but again did not

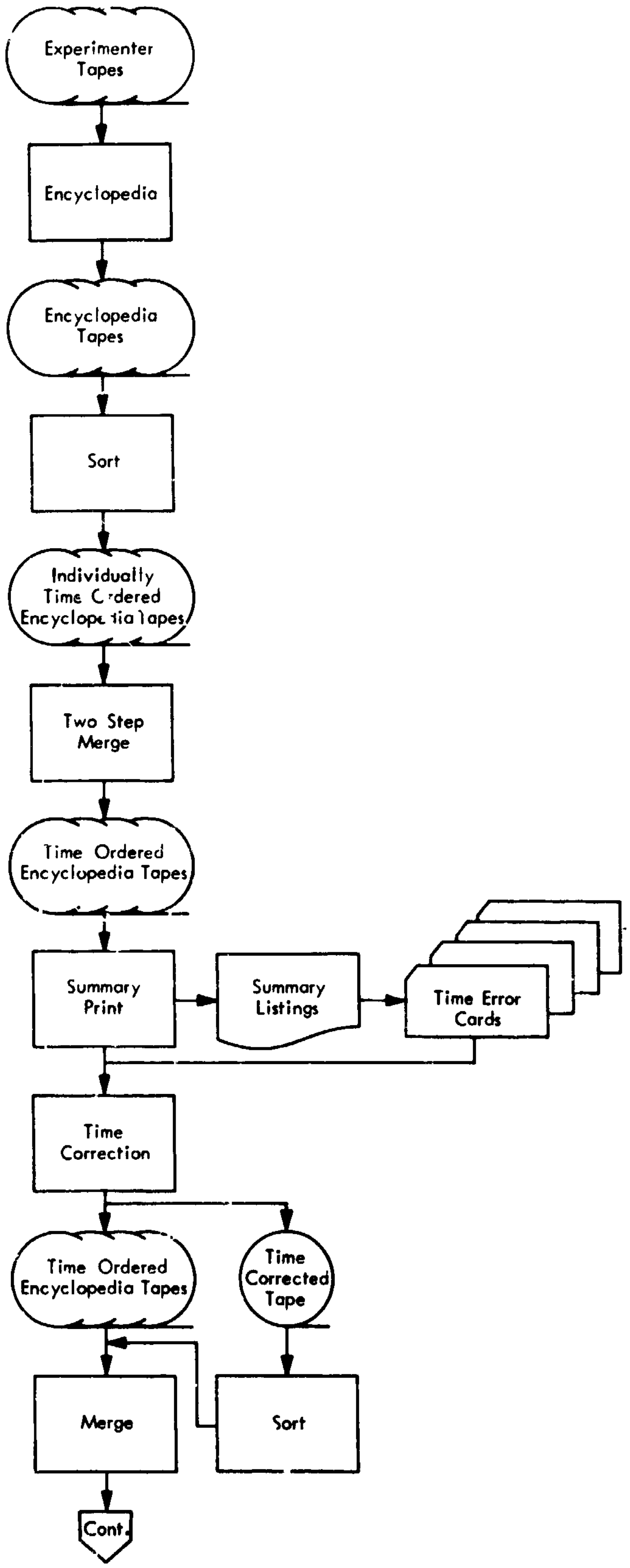


Figure 24. S3 and S3A Cosmic Ray Data Processing System (Sheet 1 of 2)

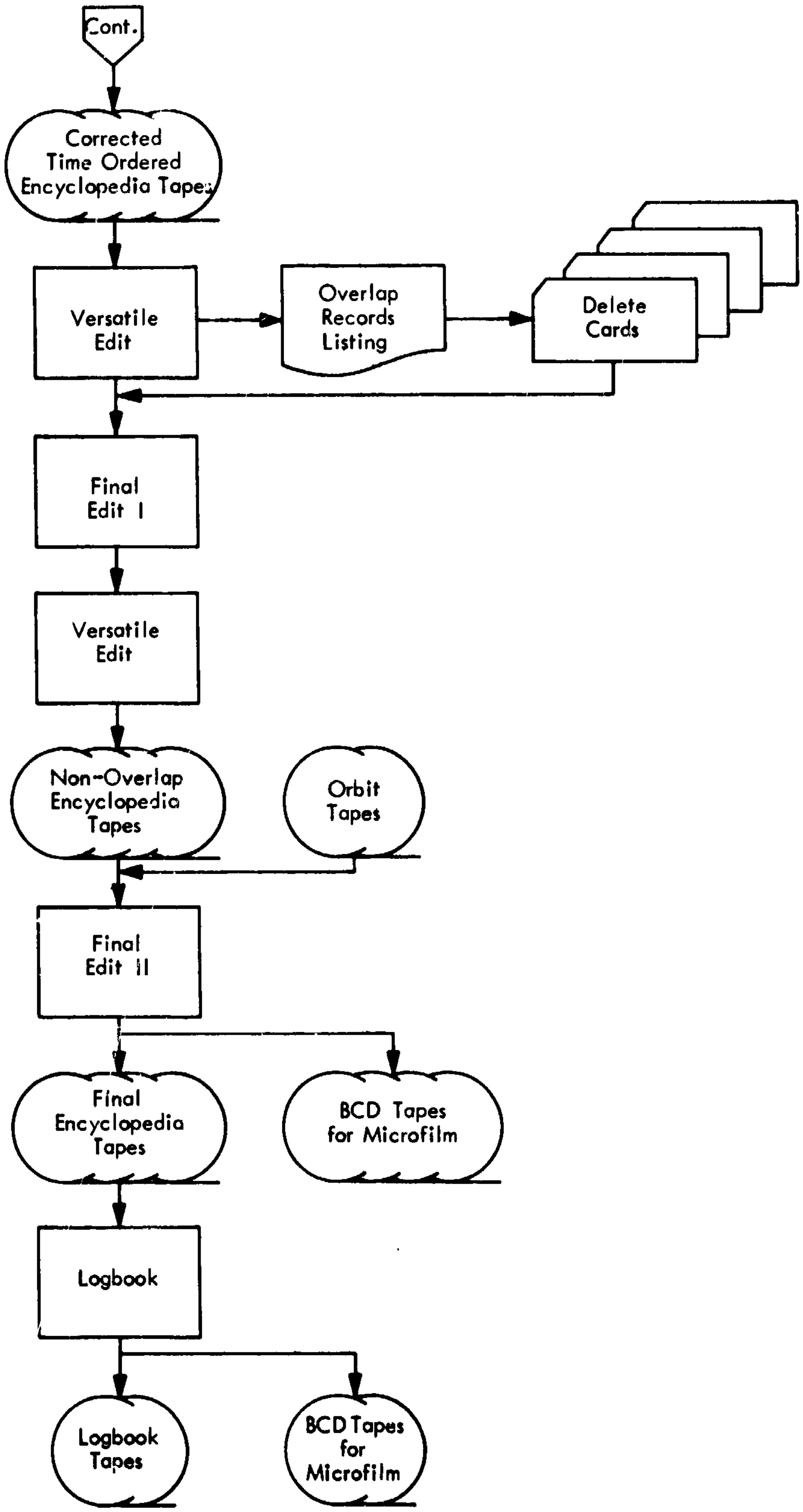


Figure 24. S3 and S3A Cosmic Ray Data Processing System (Sheet 2 of 2)

function correctly. Modifications were suggested to the division and another modified routine was successfully tested.

The ENCYUD program was successfully tested for both S3 and S3A data. The summary tapes produced were used to test the modified LGBKUD. Both the S3 and S3A test data were successfully processed by LGBKUD. The updated Logbook tapes produced by LGBKUD were used to test S3HELP, a plotting program. See Figure 25 for data flow through ENCYUD, LGBKUD, and S3HELP. The input to ENCYUD and LGBKUD are produced by S3DPS, see Figure 24 for the S3DPS data flow. (S3HELP is described in the following section.)

Dr. Upendra Desai (NASA) had requested modifications to the microfilm copies of the S3 Encyclopedia data, in the event that they were regenerated. The approval to regenerate the microfilm was received only two work days prior to the shipping of the tapes used in the production of the microfilm. The Microfilm (MICROF) program, required to modify the tapes to be microfilmed, was written and successfully tested. The input tapes to MICROF are the updated Encyclopedia tapes produced by ENCYUD. All the S3 Encyclopedia tapes were then processed by ENCYUD to produce the input for MICROF. The tapes were run through MICROF and checked by the Format Check (CHECK) program. Due to the priority required to obtain the necessary computer time, all programs were run with the programmer present. Through the personal effort of the programmer, the tapes were ready in time for shipment.

S3, S3A Cosmic Ray Section Analysis

Additional task modifications were received from Dr. Desai for the S3, S3A Height and Energy Level Plot program (S3HELP). S3HELP generates tapes to be plotted by a Stromberg Carlson 4020 cathode ray tube plotter (SC 4020). There are two separate types of graphs generated, an orbital graph starting at apogee, and a one-hour graph beginning whenever the level one energetic particle count is fluctuating in the 50-1500 range. (See Figures 26 and 27.)

The modifications included the separation of the two types of plots onto separate tapes. This allowed the creation of hard copy plots of the orbital graphs and microfilm copy plots of the more numerous one-hour graphs.

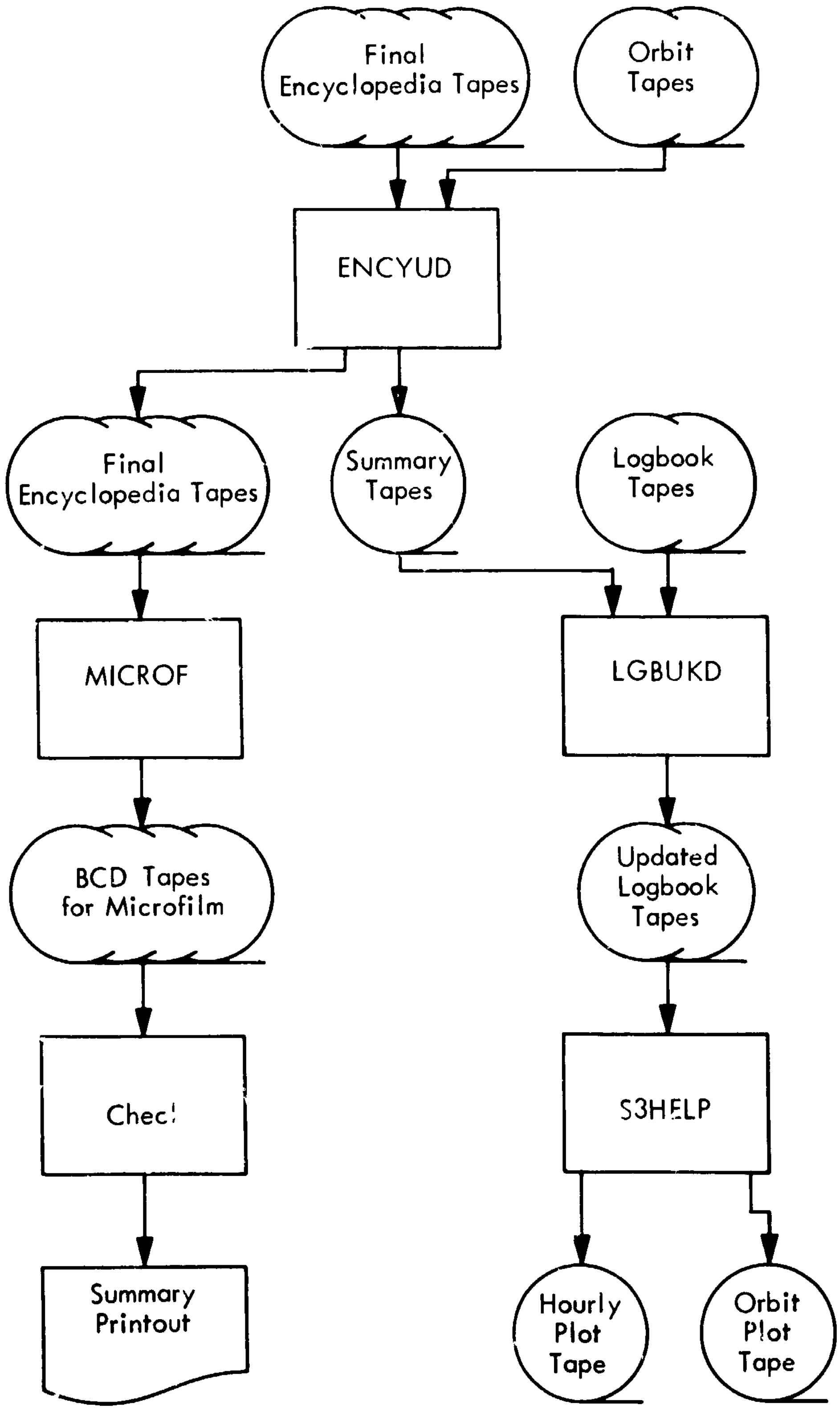


Figure 25. S3 and S3A Cosmic Ray Data Flow

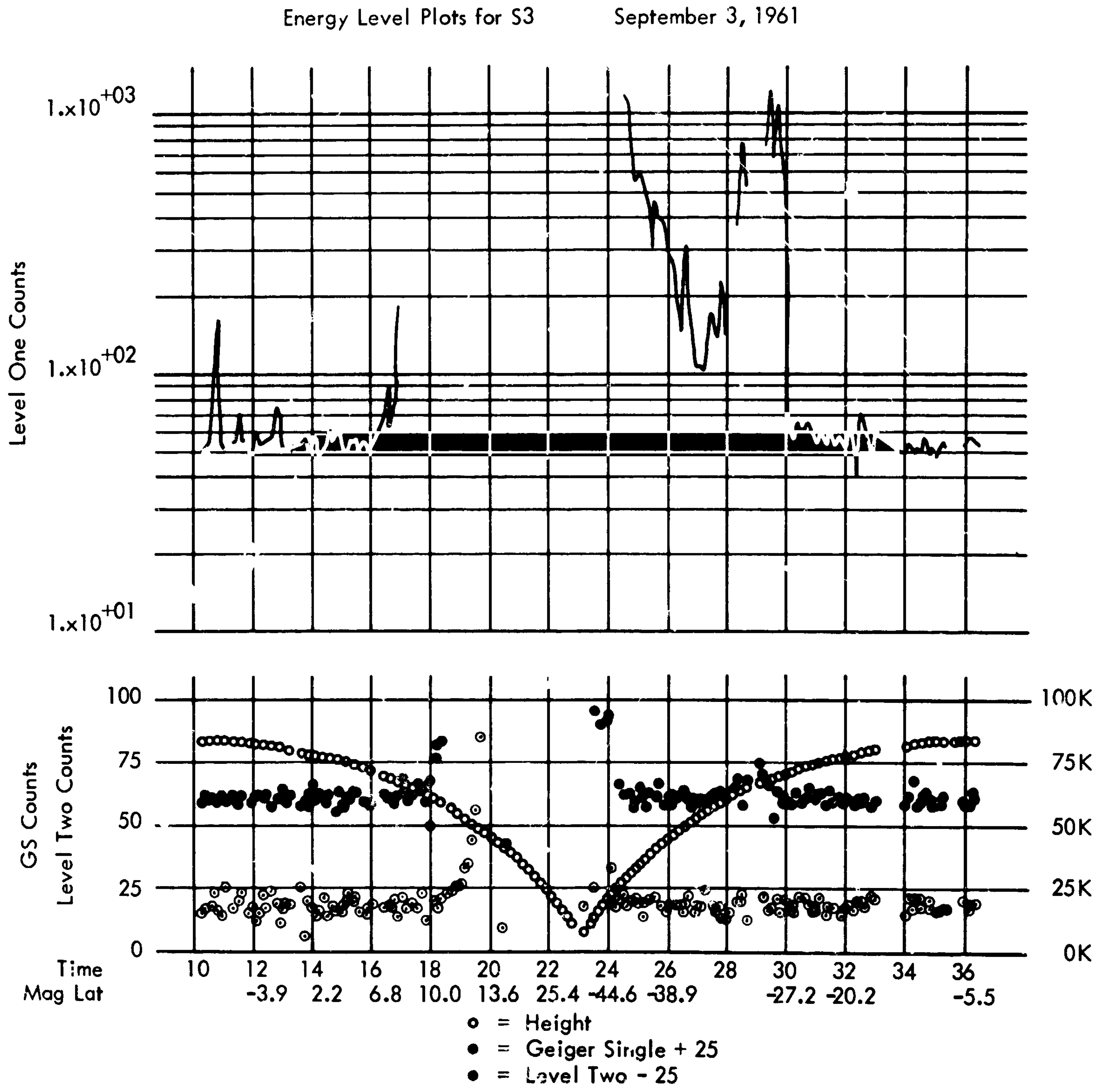


Figure 26. SC 4020 Plot from S3HELP Orbital Graph Tape

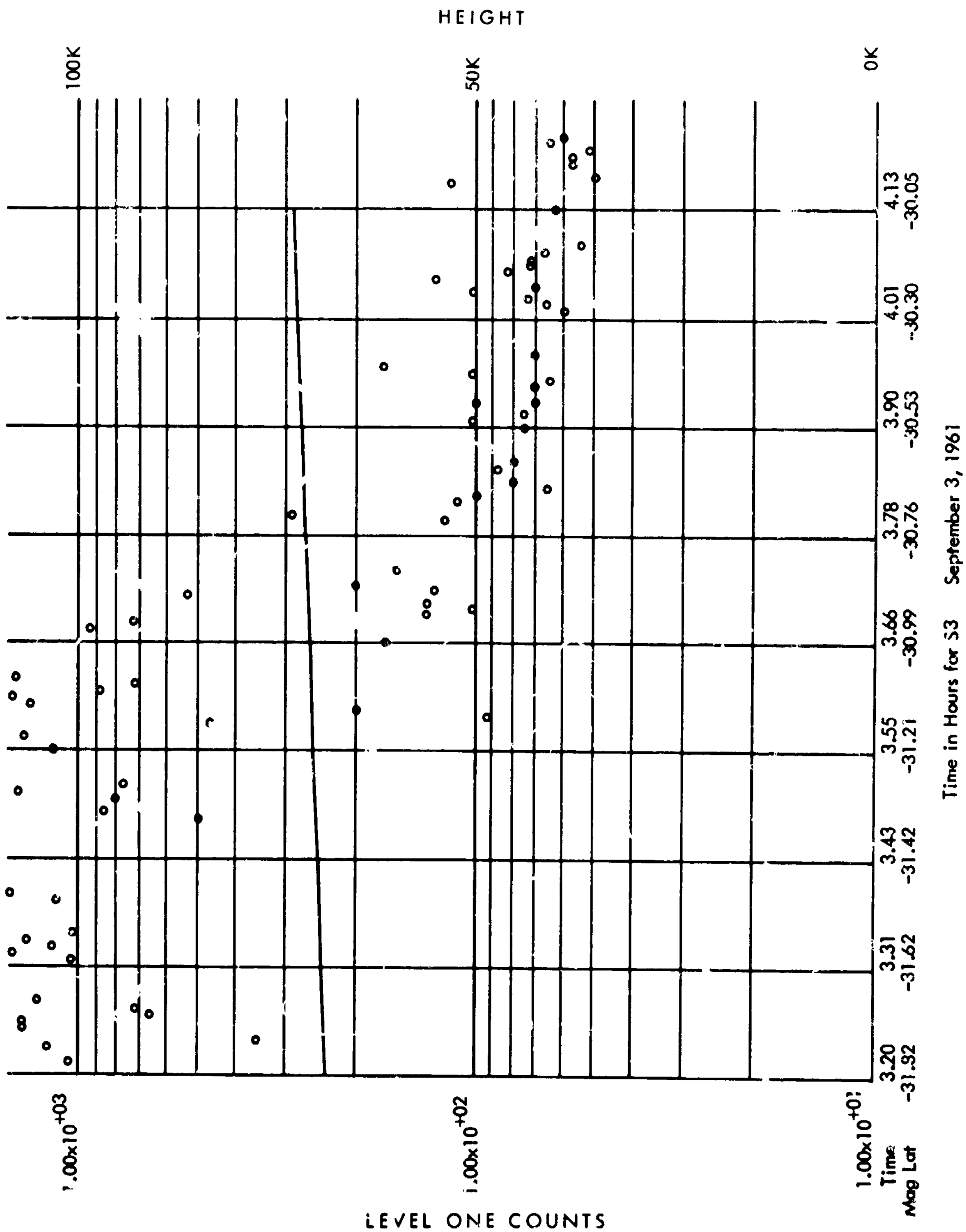


Figure 27. SC 4020 Plot from S3HELP One-Hour Graph Tape

However, the SC 4020 software support package did not allow two plot tapes to be produced simultaneously. This was learned after two unsuccessful attempts revealed that both the SC 4020 Programmer's Reference Manual and the most recent listing of the software support package did not agree with each other, nor with the actual package currently in use. A subroutine, CLEAN, was written to alter the support package at execution time. This was successful and the two tapes were generated.

At present, the plot formats have been approved for the S3 data and a modification for the S3A plot format is being tested. The debugging of S3HELP was hindered by a two-week down time of the SC 4020 and the consistencly slow turnaround time of the plotter. Hence, documentation of S3HELP was delayed.

S3C (EPE-D) Data Processing

The EPE-D Data Processing System was delivered for production during the last quarter. However, Mr. Leo Davis (NASA) analyzed the results of the first few production runs and requested several modifications to the Roll Equation Smoothing (RES) program. These modifications concerned the criteria used to determine if a smoothed roll equation was to be calculated. The modifications were incorporated into the program and released for production.

The final program in the system, MAOD (Merge and Attach Orbital Data), produces a one-line summary printout for each wheel revolution data block. Mr. Davis requested that the printout be created on microfilm, rather than the 1403 printer, as originally planned. The modifications were incorporated into MAOD and it was released for production.

The microfilm summaries are obtained from the Stromberg Carlson 4020 plotter (SC 4020). After several production runs, it was evident that the tape produced by MAOD for the SC 4020 did not always produce a good microfilm copy. This was due to SC 4020 problems and problems with the tape drives on which MAOD produced its tape. It was suggested to Mr. Davis that a program be written which could read in the final Encyclopedia tapes produced by MAOD and create a one-line summary tape for the SC 4020. This would eliminate the requirement of having to rerun MAOD (a comparatively long computer run) to regenerate a tape for the SC 4020. The plan was approved and a program was written by the technicians in charge of production, employing the coding used in MAOD.

The same technician suggested that the system should be modified to produce a system that could be run more easily through production. Modifications were made to enable the system to be executed in two computer runs, rather than five. Also, the number of intermediate summary tapes created was reduced from five to one. The changes made were mainly in control cards associated with each program, thus enabling the technician running the system to use the same object decks regardless of the system configuration used. (See Figure 28.)

The documentation for the system was delayed due to the system changes and their implementation.

S-3C (EPE-D) Attitude Determination Study

In processing several sets of data it was found that the present Phase 1 of the attitude determination system was unable to recover from the erratic behavior that the satellite undergoes at perigee. This problem was discussed with Mr. Leo Davis and the following approach was decided upon.

a. After each perigee or data dropout a special smoothing subroutine will act on the data.

b. The smoothed data will be used as input to a harmonic analysis program. This program will recover the amplitudes, phases, and frequencies of the component waveforms.

c. A pattern searching subroutine will scan the output of the harmonic analysis. It will identify the various frequencies as being precession, spin or sun-difference signals.

d. Outputs from item c will be used as starting estimates for the attitude determination parameters. These values will be used to fit the raw data to the calculated curve using GLSWS as was done before.

Work on the above modifications was begun.

Late in the quarter, it was suggested that the number of variables that enter into the calculation in a non-linear way can be reduced by two if three linearly dependent parameters are introduced. This should have the effect of improving

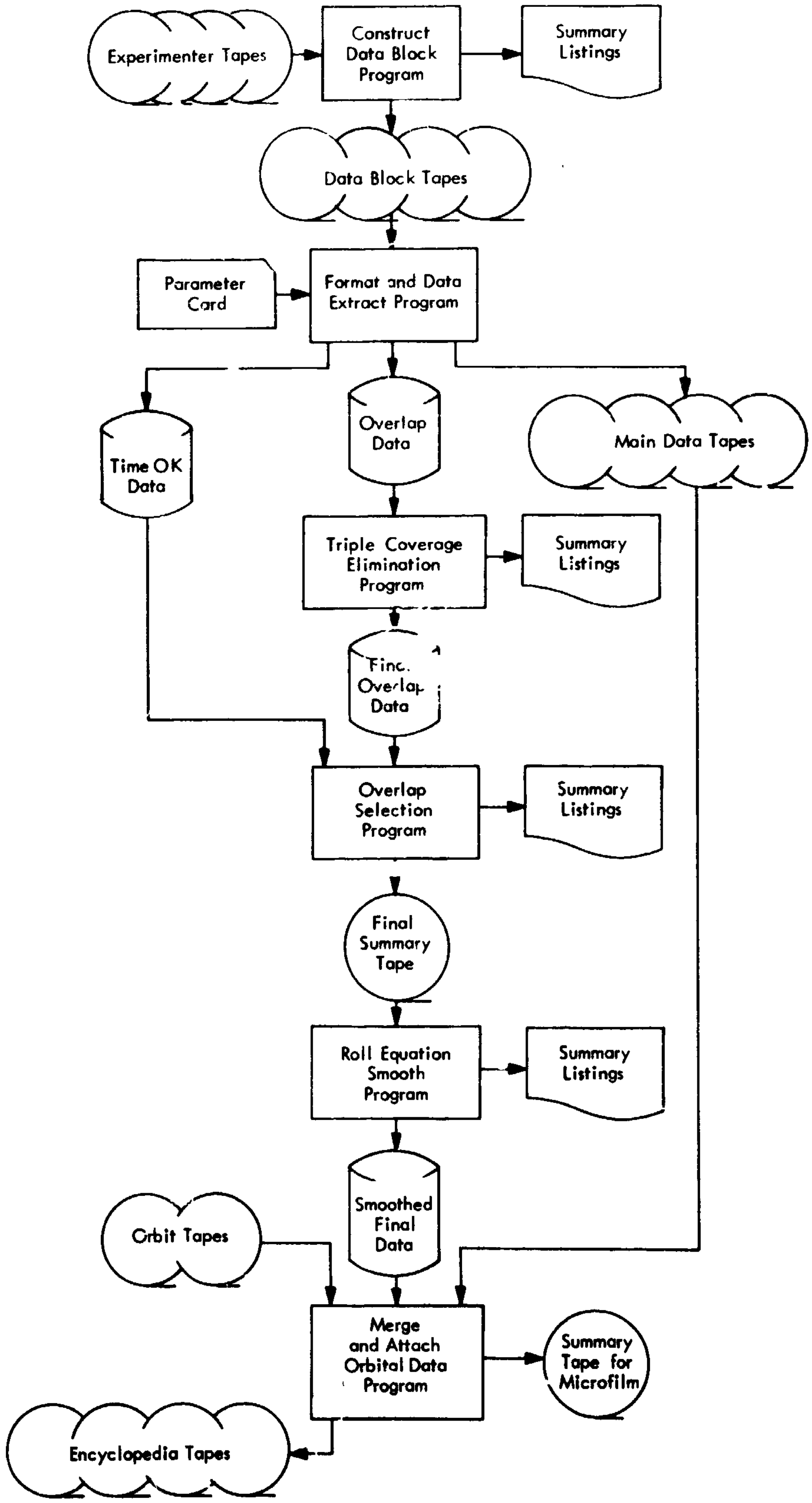


Figure 28. S3C (EPE-D) Data Processing System (Two-run Execution)

the speed of the fitting process as well as enabling convergence with poorer starting estimates.

Plots of calculated versus actual solar patch data exhibit a non-linearity. This implies that a linear calibration curve, as was assumed, may not be correct. On the basis of these plots a calibration curve was generated and the root mean square deviation of the calculated versus calibrated data showed a 30 to 35 percent decrease as compared to the same calculation using uncalibrated data.

Tests have begun to determine if this calibration curve is constant throughout the period of time of interest. If it varies, periodic recalibration will have to be included in the calculations.

EGO-A Data Processing

The EGO-A Encyclopedia program was modified to accept experimenter decom tapes directly, eliminating the need for the tapes to be blocked. Previously, the decom tapes were blocked on the 1410 computer before its removal in October 1966.

POGO-II Data Processing

The specifications for the POGO-II Data Processing System for Dr. R. Hoffman (NASA) were completed, and two of the programs in this system were written and debugged. The first program, PGPRNT, reads and unpacks data from the experimenter decom tapes and prints this information out. The records and files to be printed are selected with parameter cards. The second program, CONDENSE, reads attitude-orbit tapes, converts the parameters required for further analysis to System/360 internal representation, and writes the resulting parameters on a nine-track tape. One record is written on this tape for each orbit processed. This greatly reduces the number of attitude-orbit tapes required for further processing.

IMP-OGO Analysis

A request was received from Dr. V. K. Balasubrahmanyan (NASA) to write a program that will produce plots of Geiger Telescope Data from IMP-A, IMP-B, IMP-C and OGO-A against time on a three-hour basis. A plot of KP (geo-magnetic activity index) values was also required on the same graph. The three programs, KPTTPE, IEDAS3, and KP3HPT, required to produce these plots were written and debugged. The first program, KPTTPE, reads cards containing eight KP values per day, reformats these values and writes them on a binary tape. The second program, IEDAS3, reads the five geiger telescope hourly average print tapes for the four satellites, averages the hourly averages on a three-hour basis combining the data from all four satellites, and writes the results, along with the original values, on a binary tape. A printout of all the data processed, along with the resulting averages and the number of points used in computing these averages, is also produced. The third program, KP3HPT, reads the binary tapes produced by the first two programs and produces a plot tape for the Gerber plotter. Each plot covers a period of one month. The satellite data and the KP values are plotted with separate pens on the same plot. The scale used for the satellite data is adjusted to give maximum resolution for each set of data.

Users guides for the plot programs in the IMP Data Processing System were written in preparation for turning these programs over to data technicians for future production runs.

Conversion of the SUMOGO program to System/360 operation has been completed. The System/360 version of the subprogram BIMAT was modified to produce a second output tape which could be read by the IBM 7094. The tape is in the same format as the tapes which were generated by the IBM 7094 version of BIMAT. Minor differences in the printouts of the System/360 and 7094 versions were found to be due to errors in the 7094 version. The seven-track BIMAT tape produced by the System/360 version of SUMOGO was run through the ANALOGO program on the 7094 and the results were correct. Dumps of the nine-track BIMAT tape produced by SUMOGO were checked visually and found to be correct.

A modification of the SUMOGO/360 program was requested by Dr. Hagge. The following specifications were requested:

a. Produce separate output tapes for rates information and PHA data

b. Produce printouts and BIMAT tapes for PHA data on an orbital basis instead of a daily basis

c. Continue to produce rates information on a daily basis, but store rates information all the time rather than exclude dead time rates data

d. Calculate perigee time for an orbit, given just the orbit number rather than read in orbit numbers and their corresponding perigee times

Additional modifications are being made to provide for:

a. Eliminating unused variables and logic

b. Reorganizing existing logic

c. Changing the logic numbering scheme

d. Reorganizing the common storage area.

These changes were made to conserve storage, optimize running time and increase logic readability to aid in program understanding and debugging.

The analysis for the requested modifications and additional improvements has been completed. About 80 percent of the coding has been completed.

Gerber Plot Package

The Gerber Plot Package has been documented and a copy of the documentation has been given to Mr. Bracken of the Laboratory for Space Sciences. To make the Package more convenient to use a copy of the object decks has been placed on a tape at the DCS computer. Using the IBSYS editor any desired Gerber Plot subroutines may be inserted into the input stream by naming the desired deck. This procedure eliminates the need for carrying and handling a large binary deck.

Spark Chamber

The Spark Chamber program has been modified to process tapes received from Cornell University with spark input data. The flag which indicated the termination of a spark event was missing on the Cornell tapes and another means of detection sought. It was decided to consider the cyclic action of the grid data as a termination of one event and the beginning of another. These modifications were incorporated into the Spark Chamber program and the Cornell tapes were processed.

A program was written to plot housekeeping data from the Spark Chamber experiment on the Gerber plotter. The analysis, coding and debugging has been completed using sample data in the debugging runs. Production has not started due to experiment problems in generating correct times on the data tapes. However, the program is ready for production.

At the request of the Nuclear Emulsion Section the Lab-Module program was converted for use on the IBM 360. Considerable analysis was required to convert one subprogram because the program existed only in object deck form and no listing was available. The subprogram logic and its functions had to be determined from the calling programs.

Konradi Plot Program

The program (AKPLT) is designed to process S3 binary MERGE tapes, for Dr. A. Konradi (NASA), containing data readouts for channels 8 and 10, and then produce hardcopy plots in the Gerber-800 series plotter. The plots are drawn onto a pre-formatted sheet of vellum (see Figure 29), one orbit per file, consisting of 26 hours of data. The various sections to be drawn are as follows:

a. The Eighth Dynode Channel Readings for Wheel Positions (WP) 2 through 9, utilizing the conversion formula for all WPs, except 3

$$I_8 = 10^{\left(\frac{L-1100}{100}\right)} \text{ AMPS.}$$

for WP 3,

$$I_8 = 10^{\left(\frac{L-1007}{100}\right)} \text{ AMPS.}$$

set up on a linear scale ranging from 0 to 700.

b. Eighth Dynode Channel Readings for WP 9, and 12 through 14, using the conversion formula above.

c. Tenth Dynode Channel Readings for WP 2 through 9, utilizing the conversion formula for all WPs, except 3

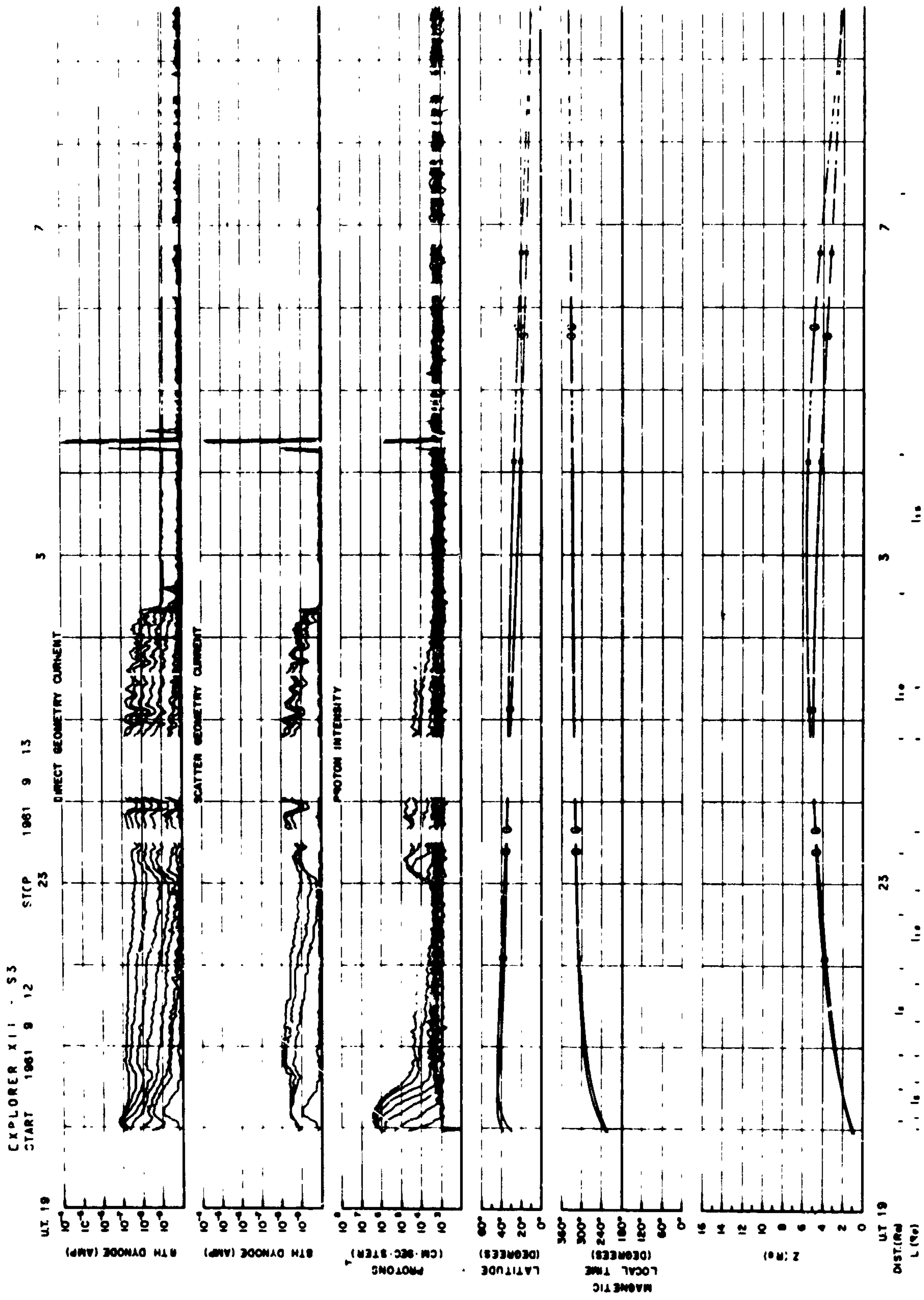


Figure 29. Sample Plot from AKPLT Program

$$I_{10} = 10^{\left(\frac{L+226.6}{100}\right)} \text{AMPS.}$$

for WP 3.

$$I_{10} = 10^{\left(\frac{L+325.6}{100}\right)} \text{AMPS.}$$

set up on a linear scale ranging from -123.5 to +576.5.

d. Latitude for geomagnetic and solar-magnetospheric systems

e. Magnetic Local Time (MLT) for the geomagnetic system computed by the difference between (Sun-Satellite) longitudes, and the solar-magnetospheric system defined by its actual solar-magnetospheric longitude.

f. Z(RE), defines attitude for both geomagnetic and solar-magnetospheric systems utilizing the conversion formula:

Z(RE) = DIST (RE) *SIN (LATITUDE)

In addition, the following items are accounted for on the plot:

a. Satellite heading with orbit start/stop time information

b. Universal time listings from the start of the orbit

c. Distance and McIlwains parameter marked in integer and tenth of earth radii intervals.

During implementation of the program, it was necessary to modify the wheel position acceptance criteria for channels 8 and 10 several times in an effort to filter out unwanted noise and data.

A further modification was needed in the conversion criteria used for plotting MLT and altitude, and a method was added to overlay and label P/N values on a positive 0^{o}-360^{o} plotting surface.

Due to the large input data arrays and the Gerber plotter input package, the program utilizes all available core locations on the DCS computer. In making modifications to the specifications, core usage had to be taken into account to prevent core storage overflow. This was remedied by removing those Gerber routines which were not being used in the plot package, grouping I/O buffers, and dividing the plot area into four seven-hour sections covering a maximum of 28 hours of data per pre-formatted sheet of plotting paper.

During the testing phases, the original pre-formatted graphs proved to be inadequate due to errors incurred by the tools which were used by the draftsmen who produced the mask. A program to provide a more accurate, one-time mask for pre-formatting was written, plotted on the Gerber, and then delivered to the draftsmen for special labeling. This mask will be copied onto vellum, on a one-to-one basis, for use in production plotting.

Further program testing resulted in the addition of a lettering routine to the Gerber plot package, rather than the normal print-wheel printing routine.

Computer problems evolved on both the DCS and Gerber machines. A number of generalized subroutines which were furnished for this task were in error and resulted in DCS core storage overflows. After a series of core-dumps, the errors were located and corrected. The plotter encountered several down time problems which delayed program system testing, and malfunctions developed with the pens and ink which were used in the plotting. Both problems have since been resolved.

The plot program is now capable of producing one full orbit of data (40,000 points) per 3.2 minutes of DCS time and 1.5 hours of Gerber plotter time. By means of building arrays in core containing locations of the data points for both X and Y values, rather than the MERGE data itself, core storage was conserved and plotting time reduced.

Production for this task will be handled by Dr. Konradi. There will be 102 orbital plots produced, utilizing 54 hours of DCS time and 150 hours of Gerber plotter time.

Balloon Flight Data Processing

The modifications to the Balloon Data Processing System were completed and tested in January. Operating procedures were written and the system was turned over to Vitro Laboratories, Inc., for production. At that time, although these were only five basic programs in the system, there were as many as four versions of some of the programs in production. This was due to the fact that there were different formats on the input tapes and at times it was required to process only high gain or low gain data.

In March, it was discovered that several of the programs were inadvertently skipping over one in every hundred records. While this malfunction was being corrected, it was decided to consolidate the various versions of the program into a single program. As a result, the system now consists of six object decks as opposed to fifteen before. The procedures for setting up jobs has been simplified, there are fewer decks to maintain and the response time for future modifications should be considerably reduced.

A request was made by Dr. Balasubrahmanyan that an Extended Background program, capable of producing 80 by 80 matrices, be written. The program was coded, debugged and placed into production.

A program to perform a least squares fit on two short sets of data was also written and delivered.

Power Spectrum Analysis

Several small modifications were requested and made during the early part of the quarter. All documentation was completed thus completing work on this subtask.

Generalized Tape Manipulation Subroutine - FILE

A task request was received to provide a subroutine which would backspace over and read forward over files on a tape mounted on any FORTRAN logical unit. FILE was released for production use and the documentation has been completed. Several programmers and scientists have successfully used the routine in existing programs.

Program EXTRA

After completing seven hours of production runs on the IBM System/360, it was decided to further analyze and, if possible, optimize the EXTRA program, because at the current time rate it would have been necessary to use 44 more hours of computer time to complete the task.

Some improvements have been incorporated into the program and tests performed to ensure the accuracy of the results. Further modifications are being studied to reduce the amount of computer time required.

Nuclear Cascade

Several modifications to the Nuclear Cascade task were received during the quarter:

a. The task was expanded to include another major program which will calculate the losses that electrons suffer in passing through successive layers of scintillator and absorber. The program ELK (Electron Shower calculation) (see Appendix A) will use a Monte-Carlo technique which will require the running of thousands of test cases. Coding of the program has begun.

b. A request was received to include a program to calculate certain absorber information for different types of absorbers. Results of this calculation are needed for the ELK program.

c. A request was also received to make a modification in the method used for the Nuclear Cascade calculation. Integration was performed by a subroutine. The new method eliminates the use of the subroutine and also stores intermediate results that are useful for later integrations in the program. Tests indicate that the new method produces an increase in speed by a factor of 15 with little loss of accuracy.

Modification in the output format was also requested and completed. An additional modification has been requested which will require the production of graphic output as well as printed output. Work on this modification has been started.

PROGRAM FOR NEXT REPORTING INTERVAL

System testing of the LGBKUD and S3HELP programs will be completed and the processing of the S3 and S3-A tapes through these programs will also be completed. All documentation for the S3, S3A Cosmic Ray Data Processing System will be completed and published. A study will be undertaken to determine the best method for modifying the S3-C Data Processing System to enable it to process S3-A and S3-B data tapes. Assuming no major bottlenecks, the modifications will be made and processing of the data tapes will begin.

System testing and documentation of the POGO-II Data Processing System will be completed.

The SUMOGO-B program will be converted to process data on an orbit basis rather than daily as is presently done. The ANALOGO and SOLOGO programs will be converted to System/360. Any changes in the ANALOGO program, required as a result of the modification to SUMOGO-B, will be incorporated.

Production runs through the Spark Chamber Housekeeping Data Plot program will be set up as soon as corrected data tapes are received. Documentation for all Spark Chamber programs will be published.

The Nuclear Cascade program should be put into operation after the graphic output section is coded. The Electron Shower Calculation (ELK) program should be ready for testing early in the quarter.

Task 16-A

GENERALIZED SATELLITE DATA ANALYSIS SYSTEM

DISCUSSION

Motivated by an increasing interest in the Generalized Data Analysis System (GSDAS), the scope of the task was expanded significantly. This led to the formation of a NASA working group composed of potential system users to direct design efforts by providing IBM with functional requirements. The group is headed by Mr. Larry Hyatt and includes experimenters from the Laboratory for Space Sciences and Dr. Vette from the National Space Science Data Center.

The expansion of GSDAS' scope occurred in two areas:

1. The number of GSDAS services was increased.

2. GSDAS is to be designed for System/360 and the 7094.

Since GSDAS is still in the early design stage, an analysis of differences between GSDAS-360 and GSDAS-7094 was not required. Work was, therefore, concentrated on redesign of system concepts, published in December 1966 (Ref 4), in view of the increase in the number of GSDAS services. The following is a brief description of current design status and problems. It is emphasized that the design is still subject to changes as additional detail is developed.

General Description

GSDAS is designed to assist the experimenter in handling data files of different formats and to maintain organized data bases with a minimum of programming. The primary functions performed by the system are:

a. I/O operations for any data base whose form and structure can be defined to the system

b. Preliminary overlay analysis to determine potential data redundancy (Actual redundancy determination will be left for a user coded routine.)

c. Ability to reference data items by name without regard to format or location of the item in a data record

d. Automatic maintenance of data record sequence and data set sequence within a data base if specified in the user description of that data base

e. Cataloging of data for ease in retrieving a particular sequence of data

f. Ability to create data sets with macro instructions, thus eliminating the need for user programs to deal with data structures or output programming

g. Automatic calculation of time for every variable in a record based upon start time of the record and time factors given in the directory for every data item

h. System maintained communication between user subroutines

System Use

The GSDAS functional flow is shown in Figure 30.

To process data within GSDAS, a procedure is written consisting of GSDAS language statements which may reference machine language subroutines and other compiler language routines. This procedure definition will be processed by the GSDAS translater, tested for consistency and completeness, and translated to a compiler language and/or machine language for compilation. In the course of translation from the GSDAS language, the data base directory and the storage medium information in the catalog will be used to set up all the I/O code to handle data elements and to plan and set up the processing required to maintain the output data base in the structure defined. When the procedure calls for a subroutine, the subroutine library will be used to determine the working area required by the subroutine, the form of the variables to be stored in that area, and the location and form of the code for the subroutine.

Next, if the data sets being input are cataloged, the catalog will be searched to determine the location of the data to be read in. If it is not on DASD (Direct Access Storage Device), a message will be sent to the operator

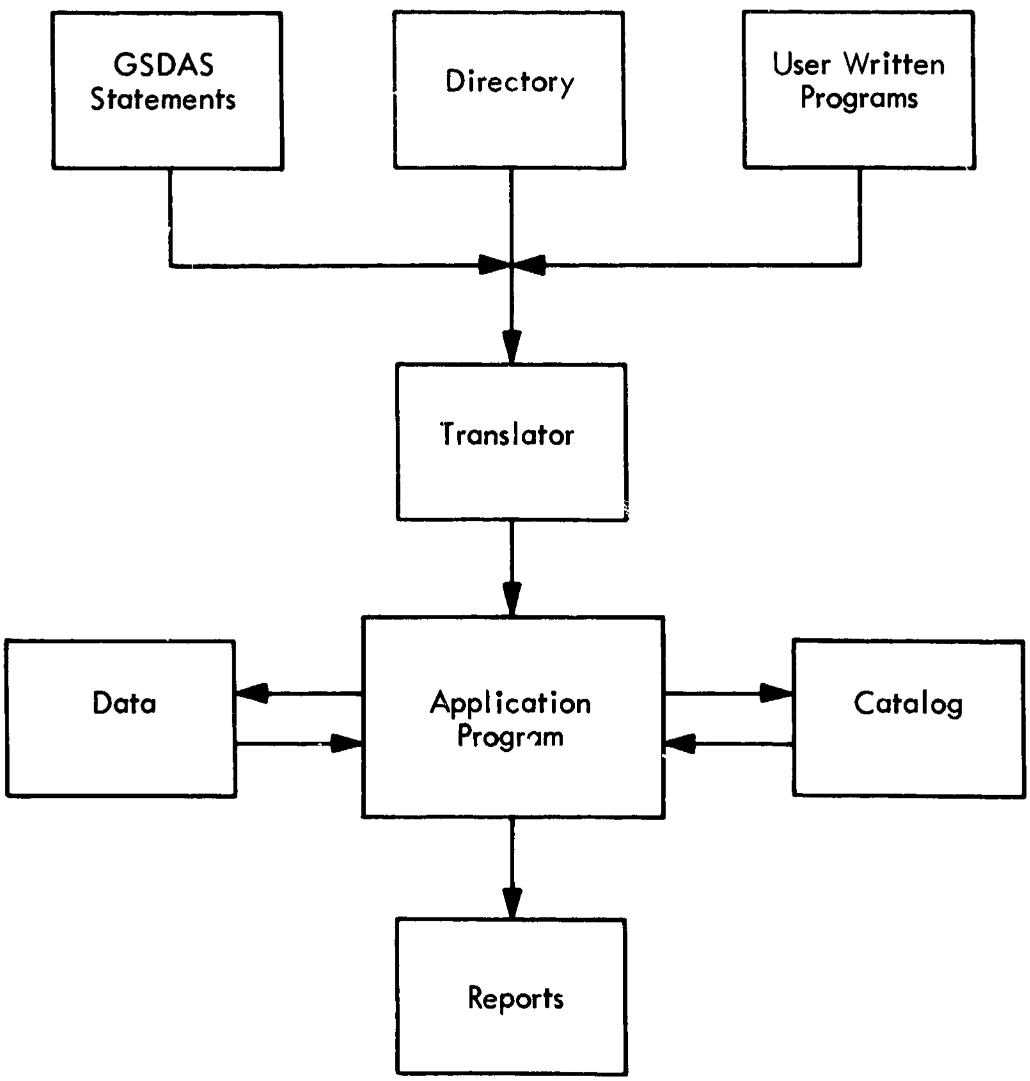


Figure 30. GSDAS Functional Flow

identifying the tapes, data cells, or disk packs to be mounted. With the data base mounted, the processing begins.

If data is being created for a cataloged data base during the procedure, preparation will be made for sorting and merging with existing data. Sorting, merging, and updating of the catalog will be performed automatically for the user. If redundancy elimination is also desired, a user subroutine (which must be on file in the system) will be invoked to make decisions on redundancy and/or data errors.

Language

The purpose of the GSDAS language is to provide a mechanism for:

a. Referencing data by name and other attributes (such as time of observation), without concern for location and format of data

b. Simplifying passing of data between subroutines

c. Simplifying creation of data sets and logical records

d. Simplifying the task of specifying data base maintenance procedures (such as sorting, merging, elimination of redundancies, etc.)

e. Converting data base formats

The syntax and vocabulary of language statements satisfying these objectives is being investigated. GSDAS statements will be similar (but not identical) to FORTRAN or PL-1 statements. A GSDAS Translator will convert these statements into statements which are interpreted by existing compilers to produce an object deck.

Catalog

As stated in the preliminary specification of December 1966 (Ref 4), the catalog will maintain a record of tape reels for each data base. In addition, the expanded scope requires the maintainenance of a record of data bases which are stored on data cells and/or disk packs. Also, an index of catalogs and data base names will be maintained.

Directory

Every file which is to be read or written by the GSDAS system must have a directory describing the format and attributes of the file. The directory will contain a unique name for "cataloged" and "to-be-cataloged" files and names for each element or array within each data record. All data records in a file must have the same format. Contiguous data elements or structures may also be named. This will enable a program to call for a single element or a set of elements with a single name. The program can then operate on the set. For example, a data record may contain three fields—year, day, minute—each of which can be called and processed. In addition, by naming the three fields—for example, TIME—the program can call for TIME, compare it with other data and move it to an output area without referencing the individual fields within TIME.

In addition to naming each data element, the directory will define the length of the item, its position in the record, and its form (e.g., BCD, binary integer, packed decimal, single-precision floating point).

Some of the data bases have a header as the first record of each data set. It will be necessary to indicate this in the directory so that when the system reads the first record of a file, it will unpack and process it as a header and not as a data record. The directory should accommodate a definition of the contents of headers, which are analogous to the content definition for a data record.

Data Base Structures

The two basic types of data base structures are single data set data base or multiple data set data base.

For the multiple data set data base the following sequencing combinations are allowed:

a. No sequencing

b. Sequencing within data sets only

c. Sequencing between data sets on time of first logical record in the data set (No sequencing within data sets here implies the user must be particularly careful that the time of the first logical record is the earliest time in the data set.)

d. Sequencing both within and between data sets

No elimination of redundant records is possible with the above structures. Moreover, the user is responsible for retaining the identify of the data set during processing by using appropriate language statements to end data sets and to open new data sets.

For the one data set data base, the following sequencing is permitted:

a. No sequencing (in which case no elimination of overlap is possible)

b. All records are in sequence (In this case, if the sort key is time, elimination of redundant records is an option.)

System Communication

Subroutines must be defined to the system as follows:

a. Subroutine name - this will be the name used in an EXECUTE statement within GSDAS language when the subroutine is to be invoked.

b. Parameter list - this must give a dummy name to every parameter required by or output by the subroutine. It must also specify the form of the parameter and whether it is an input, output, or both.

This information is stored in the subroutine library and must be present before a subroutine can be used in GSDAS.

To call a subroutine within a GSDAS procedure requires an EXECUTE (subroutine name), followed either by a list of actual data names corresponding to each dummy variable in the subroutine definition or by a LIST, reference to which points to a list in the procedure giving this information.

Example 1: EXECUTE SR (P1, TIME, NAME)

Example 2: EXECUTE SR (LIST3)
LIST3 (P1, TIME, NAME)

The system will set up a working area for the input and output variables and, for every variable marked as input in the parameter list, it will get the current value of the variable (either from the input data or from the output of another subroutine) and convert it to the form specified in the parameter list. At the exit from the subroutine, the variables marked as output will be stored in fields of that name.

Case Study

As an illustration of programming complications caused by lack of standardization of data formats, processing of the variable "TIME" is described below since it impacts many of GSDAS's services. Normally, time is a multi-field structure which gives, for example, year, day, millisecond. Time must be handled as a numeric quantity to perform such functions as determining potential data redundancy (overlap) and computing time for each data element in a record, based upon the time of the first element and a factor for each subsequent element. To retain the generality of the system, this problem can be handled best by indicating in the directory those fields which together contain time information and, further, by indicating in the directory the format of the time data (e.g., year, day, millisecond; or days, minutes, etc.). For each format, a corresponding GSDAS subroutine will be referenced to convert the time structure into a single numeric quantity.

Computer Experiments

In addition to system design work, experimental computer programs were written and run to provide answers to programming problems uncovered during discussions on system design. Experiments were performed using the Operating System/360 in the following areas:

a. Job Title Control Block use in processing multi-data set volumes

b. Multi-data set volume reading and writing

c. Label writing and processing

d. Positioning to the start of and within data sets

e. Linkages to and from subroutines

f. Volume switching

g. Specifying data set definition parameters at execution time

Results of the experiments indicate that

a. Multi-data set volumes can be processed under OS/360 with one data definition statement by reading and modifying the Job Title Control Block for the data set.

b. Standard labels can be written and used for both automatic volume checking and for supplying data set characteristics.

c. OS/360 contains the capability to automatically position to the start of a requested data set. Positioning within a data set is a user responsibility. Basic Sequential Access Method (BSAM) contains a positioning macro; Queued Sequential Access Method (QSAM) does not.

d. Automatic volume switching on input is only useful when processing single data set volumes.

e. Certain data set definition parameters can be supplied at execution time. Others, such as UNIT= and DEFER must be contained in the job control language.

PROGRAM FOR NEXT REPORTING INTERVAL

The design of a GSDAS system will continue and a preliminary functional specification will be published. Also, meetings with the working group to discuss the acceptability of GSDAS and revisions will be held.

CONCLUSIONS AND RECOMMENDATIONS

GSDAS will eliminate much of the programming effort presently required for data management. Specifically, it will obviate many programming tasks involved in:

- Data base creation
- Data base cataloging
- Sorting and merging
- Data retrieval

It is recommended that experimenters and their programming staffs consider standardizing procedures and data structures. This could lead to significant simplification of data handling and corresponding cost reduction.

For example, sync finding—now a formidable problem—could be simplified if programming difficulties were considered in the experiment design stage. Another example is the form of the variable "time." (See "Case Study" above.)

Task 16-B

IMP-F COSMIC RAY DATA PROCESSING SYSTEM

DISCUSSION

The program specifications for the IMP-F Cosmic Ray Data Processing System were completed and delivered. The proposed system is a one-pass program for operation on the System/360. The system will process the data tapes containing information from two experiments, the Low Energy Detector, by Dr. D. E. Hagge, and the Medium Energy Detector, by Dr. F. B. McDonald. The system will produce four output tapes: an edited data tape with no overlap in data, an hourly rates tape containing rates information from both experiments and two matrix tapes (one for each experiment). The matrix tape for the Low Energy Detector contains information from four E versus DE arrays for each orbit and the matrix tape for the Medium Energy Detector contains information from two E versus DE arrays.

Detailed flowcharting and coding of the system have begun.

IMP-F Data Processing System

Program specifications and flow charts have been received from Dr. Williams for his experiment onboard the IMP-F. Addenda to the specifications are to be given at a later date. Analysis of the current specification was completed and additional flowcharts were drawn. Test tapes were received, however, the data record was not the length indicated in the specifications. Corrected test tapes should be received towards the end of the quarter.

Coding of the read subprogram, IMPID, which reads and formats the data for main program processing, has been completed and a program to list the data tape has been written.

PROGRAM FOR NEXT REPORTING INTERVAL

The Data Processing Systems for the experiments onboard the IMP-F will be completed and will be ready for production in time for launch.

Task 19

PERTAPE

DISCUSSION

APMTR2 - RAE Apogee Motor Program

Program testing of APMTR2 was completed on schedule. A description of the testing procedure is given below. The method employed by the program is described in QPR-2.

Two program tests were designed and successfully completed by APMTR2. The first test consisted of integrating the equations of motion of the satellite without including the effects of thrust and perturbations. Under these conditions the motion of the satellite is reduced to simple Keplerian. For Keplerian motion, (pure elliptical motion) the orbital elements never change with the exception of the mean anomaly which can be computed from Kepler's equation. The orbital elements resulting from the integration of the equations of motion agreed to at least six significant digits with the initial orbital elements with the exception of the mean anomaly which was updated correctly as determined by Kepler's equation. The length of the integration was 20 seconds which closely approximates the actual apogee motor burn time. This test, in addition to proving APMTR2 operational, further established the accuracy of both the numerical integration procedure and the computation of the earth's gravitational effect.

The final program test of APMTR2 consisted of a comparison with APMTR. APMTR is a simplified version of AMPTR2 which computes neither the effects of the earth's gravitation nor the perturbative effects. It computes the final position and velocity vectors from which orbital elements are determined by adding the position and velocity increments resulting from apogee motor burn to the initial position and velocity vectors. To make the comparison, an APMTR2 run was made neglecting the effect of the earth's gravitation and the perturbations. The thrust table input to APMTR2 was the equivalent of a constant thrust for the duration of the apogee motor burn. This facilitated the computation of the equivalent velocity increment which was input to APMTR. The results of the two programs agreed to the same degree of accuracy to which the velocity increment was computed from the thrust table.

APMTR2 runs were also made to determine the extent to which the earth's gravitation and the perturbations influenced the final orbit resulting from apogee motor burn. The maximum observed effect of the earth's gravitation was 5 percent on the eccentricity and mean anomaly of the final orbit. Because of tolerances on the final orbital elements of RAE, this effect illustrates the need for APMTR2. The effect of the earth's gravitation on the other orbital elements was negligible. The effect of the perturbations on the final orbit was, as expected, insignificant. Specifically, it was on the order of one part in 10^7.

LNRORB - Lunar Orbit Program

Program LNRORB determines the lunar orbit resulting from satellite retrorocket fire, given the position, velocity, and time of retrorocket fire in addition to the magnitude and direction of the induced velocity decrement.

Work on LNRORB was resumed this quarter after a delay of five months. Analysis, coding, and unit testing have now been completed.

NORAD - Modified Rapid Prediction of Satellite Position Program

The work effort on NORAD has been resumed after a half-year delay. All coding has been completed but debugging and testing remain. Current emphasis is on the debugging of subroutine TINC which computes the start time, end time, and time increments for the data records on each VTPV 2 file.

PERTAPE I Conversion to System/360

The conversion of PERTAPE I to System/360 has now been completed. Test runs will all frequently used program options were made successfully and final listings, decks, and documentation were delivered to the Theory and

Analysis Office. The following is a list of the individual steps in the conversion effort which were completed:

a. Debugging of the FORTRAN IV versions of the UMPLOT Program and the SC 4020 Plot Program was completed although system difficulties with the SC 4020 subroutine package have caused some plots to be unacceptable.

b. UMPLOT (University of Michigan Plot) Program was converted to a subroutine of the Perturbations Program rather than have it overlay—as it now does on the 7094—the Perturbations Program when that program has finished execution.

c. Due to the infrequency of their use, the Calcomp Plot Program was eliminated from the PERTAPE I System, and the SC 4020 Plot Program was made a distinct program completely independent of the PERTAPE I System. The SC 4020 Plot Program will obtain plotting data from an output tape of the Perturbations Program.

d. The new FORTRAN IV version of OETRV (orbital elements to range and velocity conversion routine) was successfully tested and incorporated into the System/360 version of PERTAPE I, thereby correcting all PERTAPE I results previously in error because of their dependence upon this routine.

e. The FORTRAN IV version of TERP (sixth-order Newton forward interpolation routine) was completed. Agreement with the FAP version was to six significant digits.

f. A method of including the update node option of PERTAPE I in the System/360 version of PERTAPE I was devised, and the necessary program modifications were made and tested successfully.

PROGRAM FOR NEXT REPORTING INTERVAL

Documentation of APMTR2 and LNRORB, and debugging and testing of NORAD will be completed during the next reporting interval .

CONCLUSIONS AND RECOMMENDATIONS

The magnitude of the effect of the earth's gravitation, as noted earlier, on the final orbit computed by APMTR2 dictates that APMTR2, rather than APMTR, be used to determine the conditions necessary to achieve an acceptable final orbit for RAE. APMTR, since it does not consider earth's gravitation, is not capable of computing final orbits to the degree of accuracy required for RAE.

Task 21-A

ATS-A DATA REDUCTION

DISCUSSION

Analysis, coding, unit testing, and system testing was completed on all programs and subprograms in the ATS-A Data Reduction System. This includes the ATSAMN normal mode option (Figure 31) and the four reprocess options: unsuccessful redigitization, successful redigitization, time/ID correction, and redecommutation. Successful test runs were conducted using all available test data and the program system is fully operational for launch.

The Command Verification (COMVER) program was completed and was used to process data recorded while the spacecraft was at the Magnetic Test Facility at GSFC. The printout was sent to the EME coordinator for evaluation of the EME package.

Several requests were received for modifications to the IBM 1401 PHASE0 program. This necessitated a rewriting of the program for the IBM 7010 because the 8K core storage on the 1401 could not accommodate the required changes. The new PHASE0 program was fully checked on the 7010.

The ATS-B Solar Cell Radiation Damage Program was modified to process ATS-A data. In conjunction with this effort, 240 ATS-A solar cell calibration curves were plotted on the SC 4020 plotter, verified for accuracy, and delivered to Dr. Waddell, the SCRDE experimenter.

At the direction of the NASA task monitor, the LINETEST program was dropped as a requirement.

Unit testing of the Buffer Tape Print and Sequence Count Print programs was completed. The output of the former was used to assist in the checkout of the ATS-A A/D line and was useful in detecting a potential EME package problem where one of the experiments was out of the prescribed frequency range.

At the 14 February ATS-A Experimenter's Conference, several tape format changes were requested by various experimenters. These changes as well as some important procedural changes were implemented in the system and are documented in two changes to the original ATS-A specification.

Test tapes for the ATS-A experimenters were produced ahead of schedule and delivered to the Task Monitor for shipment.

PROGRAM FOR NEXT REPORTING INTERVAL

Pre-launch and post-launch support will be provided and the publication of the ATS-A Program documentation is planned.

CONCLUSIONS AND RECOMMENDATIONS

The ATS-A Experimenter's Conference was held on 14 February 1967. The experimenters requested that many changes be made to their tape formats, including requests for additional data. All of these changes required programming modifications which, in most cases, were made; however, some changes would have required extensive program modification and correction and, therefore, could not be implemented. Had these requests been received two months earlier, they could all have been made without difficulty and with no program code modifications. Therefore, it is recommended that the experimenter conferences be held from four to six months before launch so that all experimenter requirements can be satisfied with no reprogramming and/or redesign effort required.

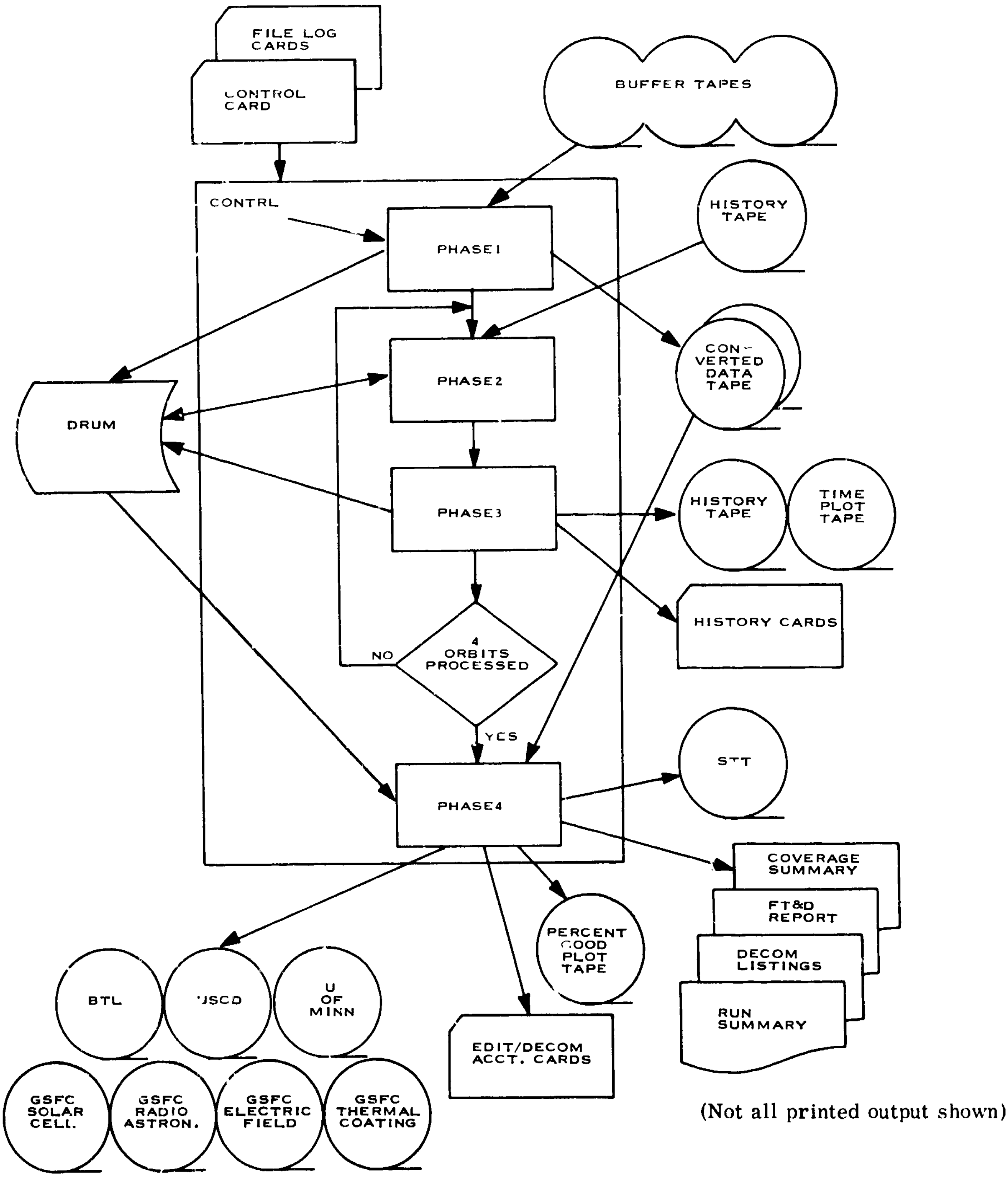


Figure 31. ATSAMN Normal Mode Processing

Task 21-B

ATS-B DATA REDUCTION

DISCUSSION

The method of automatic time analysis developed by Mr. R. R. Rosner (Ref 5) was implemented for the first time in the ATS-B PFM Telemetry Data Reduction System (ATSMAIN). The automatic time correction algorithm furnished the necessary tool to implement an accurate one pass telemetry data reduction system. This type of system provided greater efficiency and offered several distinct advantages over the multi-pass system, namely:

a. It reduced the probability of operator error by minimizing the number of operator interventions.

b. It was more efficient since the production delay, until the manual time analysis was completed, was eliminated.

c. It was more reliable since the 1301 disk is used as the intermediate storage medium instead of the more error-prone tape storage. Reliability was also increased by reducing the number of passes through the data to the absolute minimum, i.e., once on input and once on output. Previous systems have required at least two complete passes through the data.

At the close of the first quarter of 1967, sufficient ATS-B data groups have been processed to corroborate the following observations:

a. The design premise that time errors will be the exception rather than the rule has been substantiated. In fact, the sequence count-time relationship has been so stable that the automatic time correction method has been one of time verification rather than time correction, as anticipated.

b. The files with an unstable sequence count-time relationship require independent parameters to make the necessary time correction. These

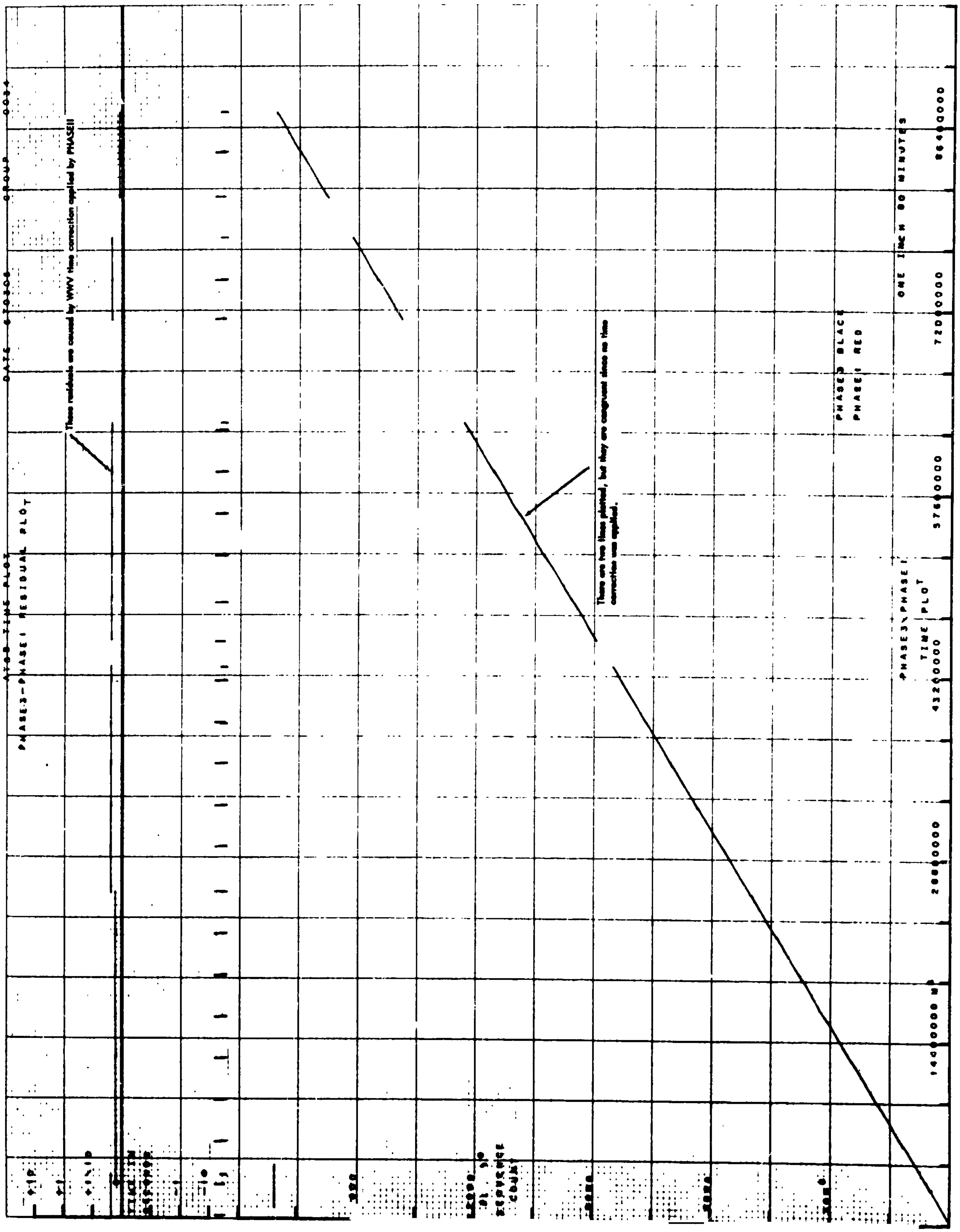


Figure 32. ATS-B Time Plot

parameters are the Analog Start Time and Analog Stop Time and are of paramount importance in effecting an accurate correction. Thus far, there have been no problems with the recording of these parameters.

c. The results of the automatic time analysis can be closely monitored using the system produced Time Plots (see Figure 32). The first plot shows grossly the results of the time corrections applied, if any. Two curves are plotted using the same axes: the sequence count-time relationship for each file in PHASEI (time is uncorrected) and the sequence count-time relationship for each file in PHASEII (time errors are corrected).

The second (residual) plot on a modified semi-log scale of the difference between PHASEIII (corrected - smoothed) time minus PHASEI (uncorrected) time versus PHASEII (corrected) time is produced above the time sequence-count plots. The PHASEIII time is calculated using the coefficients obtained from a first degree least squares fit of the PHASEII time versus sequence count. Consequently, the residual plot has served as an excellent tool for Quality Control to closely monitor time consistence and stability. Deviations to the nearest millisecond can be rapidly identified by inspection.

Rigorous examination of the output from approximately 55 groups of data processed since launch has resulted in minor changes being made to the PHASEI, PHASEII, and PHASEIV subprograms of ATSMAIN.

The PHASE0 pre-edit program was modified to produce a File Log Card for every file on the buffer tapes; however, those files for which no Analog Accounting Card exists will only be partially complete.

Minor modifications were made to the Command Verification (COMVER) and Solar Cell Radiation Damage Experiment (SCRDE) programs.

Daily morning meetings were held with Quality Control Personnel to help reach a better understanding of the significance of the statistics on the output from the ATS-B Data Reduction System. At the same time, the programmers were able to identify potential problems and correct errors with a minimal delay in production.

The ATS-B User's Guide was written and will be published by 15 April.

PROGRAM FOR NEXT REPORTING INTERVAL

Program maintenance will be provided as required and publication of the User's Guide will be completed.

CONCLUSIONS AND RECOMMENDATIONS

Most reprocessing of the ATS-B PFM data through ATSMAIN was caused by the following:

a. Quality Control personnel were not familiar with the important printouts of the system, namely:

- PHASE0 data quality report
- PHASEI data discontinuity report
- PHASEII time plot
- PHASEIV decom listings

b. In the early stages after launch, the programmers were not on the distribution list of the ATS-B Data Reduction System's output.

c. Batch processing of groups of data caused redundant errors. (A group consists of the data recorded over a 24-hour period.)

The following recommendations will eliminate most of the reprocessing:

a. Process data on a daily basis during the first month after launch.

b. For the first month, Quality Control and the programmers should work daily as a team to examine the previous day's processing results. This would serve two purposes.

1. Familiarize Quality Control with the computer output.
2. Programmers could correct program errors, if any, prior to the next day's processing as well as anticipate potential problems.

Task 22

PAGE

DISCUSSION

The specifications, unit testing, system testing, and final documentation for the PAGE Computer Program for the IBM System/360 were completed. The specifications, source decks, object decks, listing, and documentation were delivered to NASA.

The PAGE program is designed to extract schedule data from the NASA PERT "C" program output tape, update the schedules from cards and/or master data tape input, and produce a new master data tape and a plot tape for the SC 4020 display generator which will create SARP milestone charts. Also, output from the program is a listing of any input cards containing errors. The flow of data through the PAGE program is shown in Figure 33.

The details of the program operation including the input and output data formats and sample SARP milestone charts generated on the SC 4020 plotter are contained in Reference 6.

PROGRAM FOR NEXT REPORTING INTERVAL

This task terminates with the current reporting interval.

CONCLUSIONS AND RECOMMENDATIONS

The system testing phase for PAGE lasted one week longer than anticipated due to attempts to locate the cause of an error on the milestone charts generated from the plot tape produced by the program. After three days of investigation an error was discovered in the ERRLNV subroutine of the SC 4020 Plotter Subroutine Package contained in the system library. Mr. W. Foman (NASA) was notified of this problem and it is recommended that the error be corrected to avoid any additional loss of time through the use of the plotter subroutines.

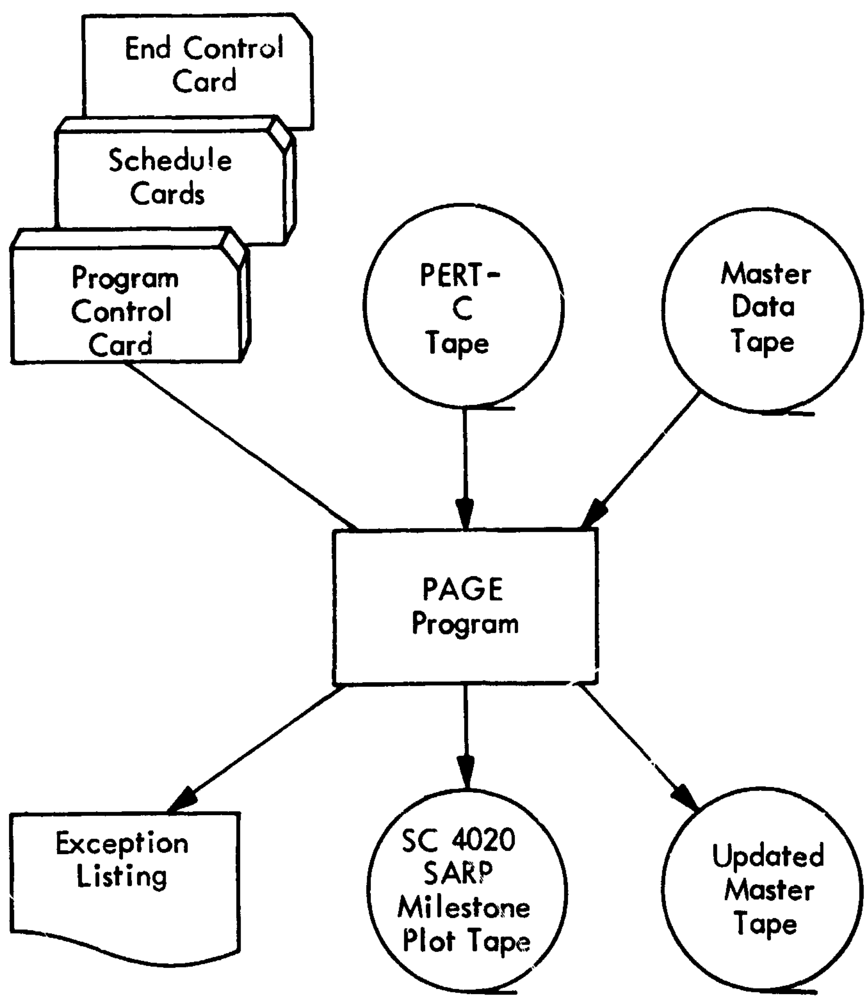


Figure 33. PAGE System Flow

Task 24

SDA DOCUMENTATION

DISCUSSION

One report "Sync Finding Techniques" was completed, reproduced, and delivered to NASA.

The organization and format of the SDA final program documentation volumes was outlined to show the various programs that will appear in each of three volumes: Volume 1, Program Systems; Volume 2, Special Purpose Programs; and Volume 3, Generalized Subroutines. Completion and delivery schedules for a total of 40 individual program writeups were prepared, projecting the work load through July 1967.

Four program writeups were prepared in draft form, edited, and sent to final typing.

PROGRAM FOR NEXT REPORTING INTERVAL

The first drafts of approximately 37 program writeups are scheduled to reach the technical editors for processing. Of these, it is expected that 30 will be edited, final typed, reviewed, corrected and delivered.

Task 25

DEFINITIVE ORBIT DETERMINATION

DISCUSSION

Subtask 1a - Specified Operational Requirements

Between 15 February and 8 March, a rough draft of the Specified Operational Requirements (SOR) was received from GSFC in four parts. The SOR was reviewed and questions and comments were sent to GSFC in three parts between 28 February and 13 March.

Although the final version of the SOR has not been received, work has begun on systems flow, input/output, data base, data file maintenance, manual input processor functions, OS/360 review, and certain processors, as directed by the rough draft of the SOR and supported by the various documents on the existing programs.

The first of a series of meetings between GSFC and IBm personnel was held to discuss various areas of concern. Meetings began on 27 March to discuss the final draft of the SOR. The next step is to specify the subset of the final DODS as its Model 1 for initial implementation.

Subtask 1b - Systems Flow

On the basis of the GSFC's SOR rough draft, an overall DODS flow diagram was drawn (See Figure 34) identifying the major subsystems and indicating the computational flow through those subsystems. These include:

- Manual Input Processor (MIP)
- Differential Correction (DC)
- Continue Differential Correction (CONTDC)
- Compare (COMPARE)
- Satellite Ephemeris Generator (EPHEM)

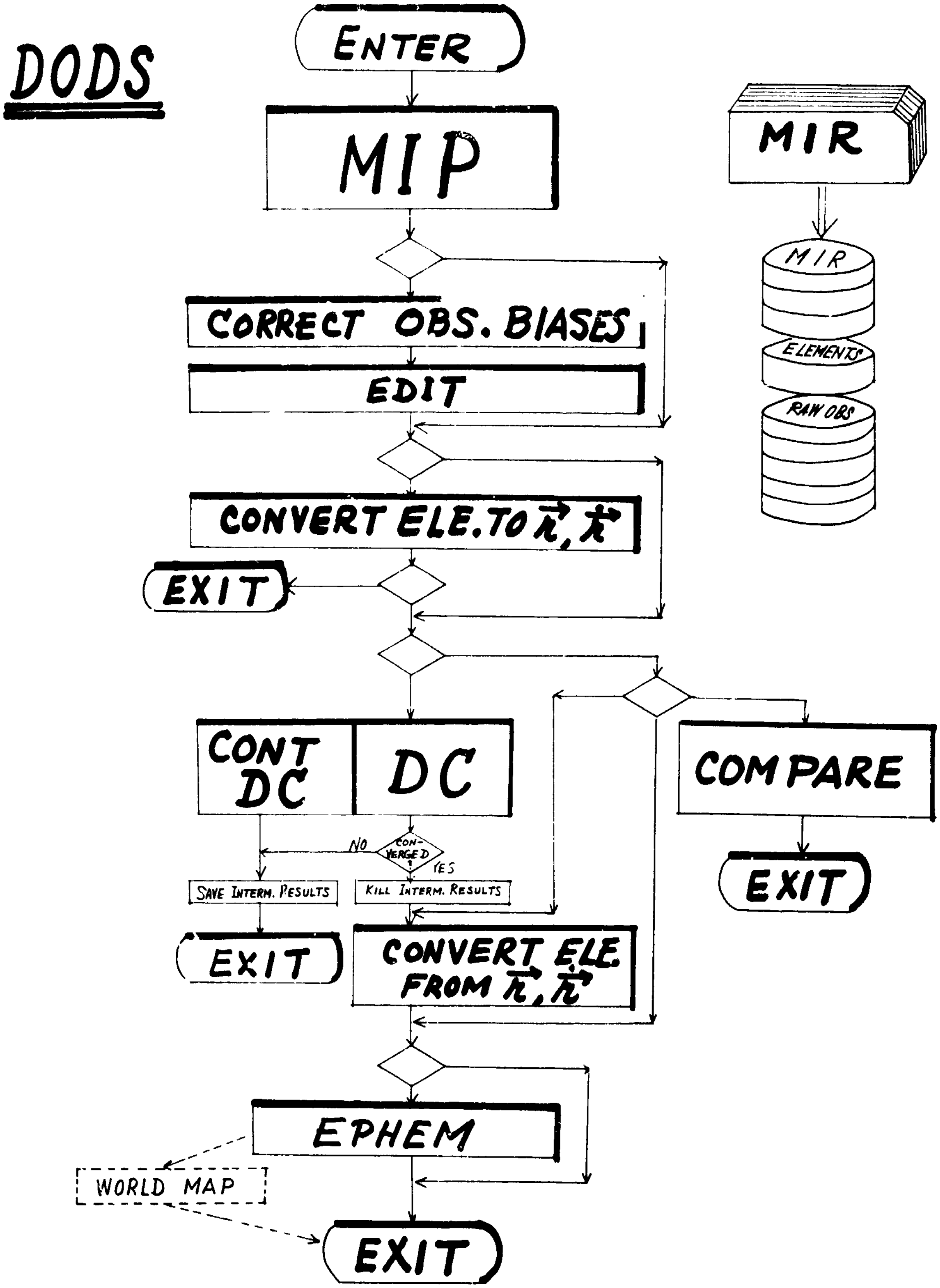


Figure 34. Overall Flow of DODS

Not shown but included in the overall design are such major subsystems as:

- Date File Maintenance (OFM, EFM)
- Data Display Processor
- System Log Processor (SLP)

A particular path through the system is defined by a set of manual input requests (MIR) by the user for specific processor(s), option(s), parameter(s), etc., which are presumably punched on cards, loaded from an on-line card reader onto a disk by OS/360, and processed by the MIP. The most commonly used paths can be stored as cataloged procedures to which the user can simply refer by names without having to punch many Job Control Language cards.

Most of the subsystems are quire large. It is evident that the Definitive Orbit Determination System (DODS) will all fit in the problem program area of the core memory simultaneously and, therefore, will have to be segmented. To solve a system of up to 60 normal equations, room for two 60 x 60 symmetric matrices, namely, about 30,000 bytes of core memory will be needed. Partial derivatives of 18 different types of observations with respect to some 300 unknowns, namely 5,400 partials, must be programmed; of these, up to 1,080 partials for the 60 unknowns will be actually computed for any one DC iteration. It is desirable to hold the matrices and the currently operating program in high-speed memory. Thus, any other data sets which may be required are best brought into core in sections as needed. This leads to the following program structure:

a. DODS constitutes one 'job' which processes one satellite at a time.

b. One DODS job is likely to be broken up in a sequence of 'job steps'; each job step being approximately the size of a subsystem of DODS.

c. The most commonly used sequences of job steps shall be stored as catalogued procedures for the convenience of the user.

d. Built-in values of all parameters and constants shall be provided in the DODS as a default option. Any of these values may be overridden by the user by means of MIR.

e. Many jobs of different options for different satellites may be batched and loaded one after another with specified priorities in an arbitrary order, waiting to be executed in the order of the priorities under the multi-jobbing environment (OS/360, option 4).

f. DODS must be so programmed that its subsystems, such as DC and EPHEM, are most efficient in the operational mode, and are most accurate in the R + D mode.

The "hands on" conversational ability specified in the SOR can be classified into two categories. One is the semi-automatic mode in which a restart with a different set of parameters and options or continuation of the computation is done after a halt. This can be accomplished by terminating a job and submitting a new job with new options. The flowchart shows how CONTDC, for instance, can be performed in the semi-automatic mode. The other is the fully manual mode, wherein the job is put into a WAIT until intervention is made from a console. Since, during the waiting period, computer resources are tied up and are not available to other jobs, use of the fully manual mode should be minimized.

A first estimate has been made of the size of the data base and it is quite large; nevertheless, it is within the capability of the Operating System/360 and the Model 75 hardware, as can be seen by the following breakdown.

By far, the largest file will be the file of observation data. About 60 bytes are required per observation. Assuming that data are kept for a two-month period on 100 satellites (minimum is 50), each having an estimated average of 100 observations per day, approximately 36,000 bytes of information space will be required. This vol me of information could be stored on one data cell of the data cell drive. Thus, while it is premature at this time to decide upon data storage, it is easy to visualize that even with a data base of this magnitude, one could easily move the information pertinent to a given satellite from the data cell to a more rapidly operational device, such as a disk or drum in preparation for a differential correction run.

The orbital elements come next. About 60 bytes are needed to contain a single set of elements for a satellite. Assuming that, of the 100 satellites in the system, one-half of them are updated daily, a new set of elements will on the average be created every second day for each satellite. To hold a one-year history in the system would thus require 60 x 100 x 1,080,000 bytes.

The ionospheric refraction index table for 400 stations during a two-month period requires 150,000 bytes. The lunar ephemeris for a similar period is perhaps 85,000 bytes long. The coefficients of the tesseral and zonal harmonics require about 4,000 bytes. Twice this amount is needed to hold the location data for 400 stations.

There are data sets required only during the DODS computation. The 60 x 60 symmetric matrix of the double-precision coefficients of the 60 normal equations is generated by the DC procedure. Two such arrays are needed to solve the equations. Storing only the upper (or lower) triangular part of the matrices still requires about 30,000 bytes. There are many other such data tables required, but under preliminary examination these do not appear to be prohibitively large.

Im summary, the demands of the DODS program for such a resource as high-speed storage appear to be satisfied with a Model 75 having over 500,000 bytes of high-speed storage. It should be remembered, however, that this high-speed core will be shared with the Operating System, System Output Writers, a Reader Interpreter, and other jobs running in parallel with DODS. Thus, in order to produce an efficient system, care must be exercised in planning the utilization of hardware resources such as LCS, the drum, disks, tapes, data cells, etc.

Subtask 2 - Documentation and Programming Standards

The following documents were delivered to GSFC on the due date of March 10:

a. Documentation Standards, including Module Performance and Design Specification, Module Performance and Design Description, and Module Flowcharting Standards

b. Programming Standards

c. Change Control Procesures

The documentation standards and programming standards form a reference handbook for the DODS personnel involved in the design and programming of the system. The documentation standards specify the format and content of the module* performance and design specification (PDS), Module Performance and Design Description (PDD), and module flowcharts. The PDS serves as the documentation standard for the DODS design effort, and as the basis for evaluating the adequacy of the completed system. The PDD, in combination with the module flowchart, will provide a complete description of the finished (and operational) DODS modules.

Programming standards have been established for DODS to ensure that a uniform self-consistent computer program system is developed. These standards will assure that (1) the many complex DODS modules will interact properly, (2) a consistent set of preferred programming techniques is used by the programmers, and (3) the completed system and its individual modules will be easily understood by others. They will be updated and expanded in the future as more knowledge is gained about the operational requirements of the system and the design objectives for DODS, Model 1 are confirmed.

The appendices of the programming standards will also be provided in the future as part of the subtask they affect. The appendices describe the details (which the main body of the report summarizes) of such standards as terminology, physical and mathematical units, module and flowchart updating, input/output isolation and formats, system parameters, and types of printouts.

Change control procedures allow for joint NASA and IBM control of changes to the system. They establish a uniform system of controls to ensure that the system meets the specified operational requirements and provide mechanisms for controlling the interactions and the effects of changes between the various modules in the system. Such procedures are required because of the complexity

*The term module will be used during the DODS effort to denote a functionally related set of instructions and data. Its development is specifically identified for documentation and monitoring by NASA-IBM concurrence. A module may be a highly complex mathematical routine (e.g., a Brouwer orbit generator) or a simple utility routine (e.g., a vector dot product module or a linkage control module). A module is also separately compilable, i.e., can be converted (from source language) into machine language independently. Thus, DODS will have executive module, I/O modules, mathematical modules, etc.

of DODS, the need for the concurrent development of DODS modules to meet the implementation schedules, and to ensure that all modules will fit together smoothly during the implementation phase.

Subtask 3 - Math Processors

Development of the DODS basic mathematical library, the orbit generator, and the differential correction processor is under way. Presented below are some details of that development.

The writing of functional specifications for the mathematical portion of the DODS system is being developed in three phases. The first phase consists of identifying and developing the basic mathematical library. Included in this phase is the definition of standard computational units, mathematical symbology, coordinate systems, and the force model.

An effort is being made to identify the functions of the Orbit Generation Processor and the Differential Correction (DC) Processor. The program identities are then used to establish a program development sequence. Identification of other DODS mathematical requirements must also be pursued.

The second phase of development consists of writing the functional specifications for those programs which are most easily defined based on current knowledge. These are listed as "unmodified processors" on the milestone chart.

The third phase of development consists of writing the functional specifications for those programs whose requirements are now nebulous. The writing of functional specifications for this group of programs will take place as the requirements become known. These are listed as "modified processors" on the milestone chart.

The following paragraphs present a more detailed discussion of the mathematical library, the force model, and the orbit generator.

DODS Mathematical Library

The DODS Mathematical Library will consist of subprograms required by the DODS Mathematical Processors and belonging to one of the following categories:

- Standard mathematical functions such as the SIN function.

- Subprograms having simple computational requirements, such as a Julian date to calendar date subprogram.
- Mathematical or statistical subprograms having wide application in engineering and science, such as a vector package subprogram.
- Subprograms which solve a frequently encountered elementary problem in celestial mechanics, such as the conversion of position and velocity vectors to orbital elements.

A list of subprogram functions to be contained initially in the library has been assembled and functional specifications are being written. The list consists mostly of functions which now exist in the present Definitive Orbit Package. The list will undoubtedly be subject to considerable additions and deletions as the requirements for library become better defined.

Force Model

The basis of the COWELL orbit generator is the force model, which is a mathematical representation of all the significant forces acting on an orbiting satellite. In general, the characteristics of the orbit and the satellite govern the significance of a given force. For this reason, the force model must be general in that it includes all forces which may have an effect on one or more satellites; yet, it must be flexible to the point of tailoring the force model for a particular type of orbit or satellite. The generality of the model will be ensured by including all forces that affect the motion of a satellite to a degree which is observable with the present observational equipment; i.e., perturbations which are not obscured by the noise in the system. Flexibility will be achieved through the use of input parameters, program option and modularity. The input parameters will include such things as astrodynamic constants, satellite descriptive constants and option control parameters. Program options will allow the exclusion or inclusion of any force from the standard model, a choice of atmospheric models, and the use of a standard force model. Modularity will make it possible to make gross changes in the mathematical definition of any one of the forces. The use of an entirely different force package may be the rule rather than the exception in the case of velocity impulse, due to the large variations in the design and use of the rockets and thrusters on different satellites.

The mathematical expression for the force model in its most general form will be three non-linear, second order differential equations representing the three components of the force vector in the inertial geocentric coordinate system. The forces to be included in the model are:

a. Gravitational attraction of the earth - The earth's gravitational field will be considered in its most general form as a spherical harmonic expansion. The series expansion will be considered out to a maximum n,m = 30 for the zonal harmonics, J_n, and tesseral harmonics, $J_{n,m}$. Options will be provided to input the harmonic coefficients and to specify a particular subset of the harmonics to be included or excluded from the model. It will also be desirable to have an automatic selection of the subset of harmonic coefficients to be used based on expected accuracies in the observational data and the orbit characteristics.

b. Atmospheric drag - The standard drag equation will be used with atmospheric densities computed from one of many possible drag models. The first system will have available a model which will account for the diurnal bulge. Means of inputing and/or computing the satellite attitude will allow a more exact calculation of the cross sectional area for irregularly shaped satellites.

c. Extra terrestrial gravitational attractions - All extra terrestrial bodies taken into consideration, initially including sun and moon, will be modeled as point masses. The design will make it possible to input necessary information to model up to four distinct bodies.

d. Solar radiation - This force will be expressed as a function of the instantaneous solar radiation pressure. The solar radiation pressure will be expressed either in tabular form or as an analytic expression which gives the pressure as a function of distance from the sun, solar activity, and elevation of the sun above the horizon. Therefore it will be necessary to enter data concerning solar activity on a daily basis. As in #2 above, since cross sectional area is important in computing this force, provision should be made to input or compute attitude of the satellite or both.

e. Velocity impulse - Changes in satellite velocity due to rocket or thruster impulses over finite time spans will be modeled with the ability to handle up to 10 such maneuvers in any trajectory. Thus, means must exist to input thrust profiles as specific impulse and mass variation tables. Also it will be necessary to input or compute the attitude of the rockets or thrusters to properly model the force. Due to the varied configurations to be expected for different satellites, this portion of the system should be modular and dependent on only the most general of input to facilitate its replacement with a minimum of effort.

The force model package, then, will be designed to ensure a maximum of generality and flexibility controlled by the operator or analyst primarily through input parameters. It will include all forces deemed significant, modeled in a manner to utilize all available spacecraft and orbit information such as spacecraft attitude. In normal operational mode a standard model will be used requiring a minimum of input parameters.

Orbit Generator

The orbit generator is defined in terms of a set of functional algorithms (Figures 35 and 36). These functional algorithms are, in turn, extended and enlarged into distinct program definitions. Before enumeration of the programs, a discussion of the basis upon which the initial definitions and subsequent extensions have been made will be presented.

Two sources provided the specifications which were necessary to define and extend the functions of the orbit generator. One was the preliminary GSFC specified operational requirements (SOR) document. The other was a group of assumptions relating directly to the task of orbit generation. All specifications in the SOR document pertaining to the orbit generator were incorporated. They are as follows:

a. The Cowell orbit generator is to be used. It will incorporate a Gauss-Jackson predictor/corrector technique, and a modified Runge-Kutta starter.

b. The capability of integrating forward and backward will be included.

c. An ephemeris which includes vehicle maneuvers, as well as the free flight portion of the trajectory, will be produced.

d. The ability to vary either the order of the Cowell formulas or the size of the integration step, or both, will be available. This will be done by input parameter, or dynamically by the orbit generator.

The assumptions are an outgrowth of several activities and considerations, some of which are: conversations with GSFC personnel; investigations into related, current analytic GSFC techniques; and the formulation of the various algorithms with modularity a prime concern. This last item will afford the

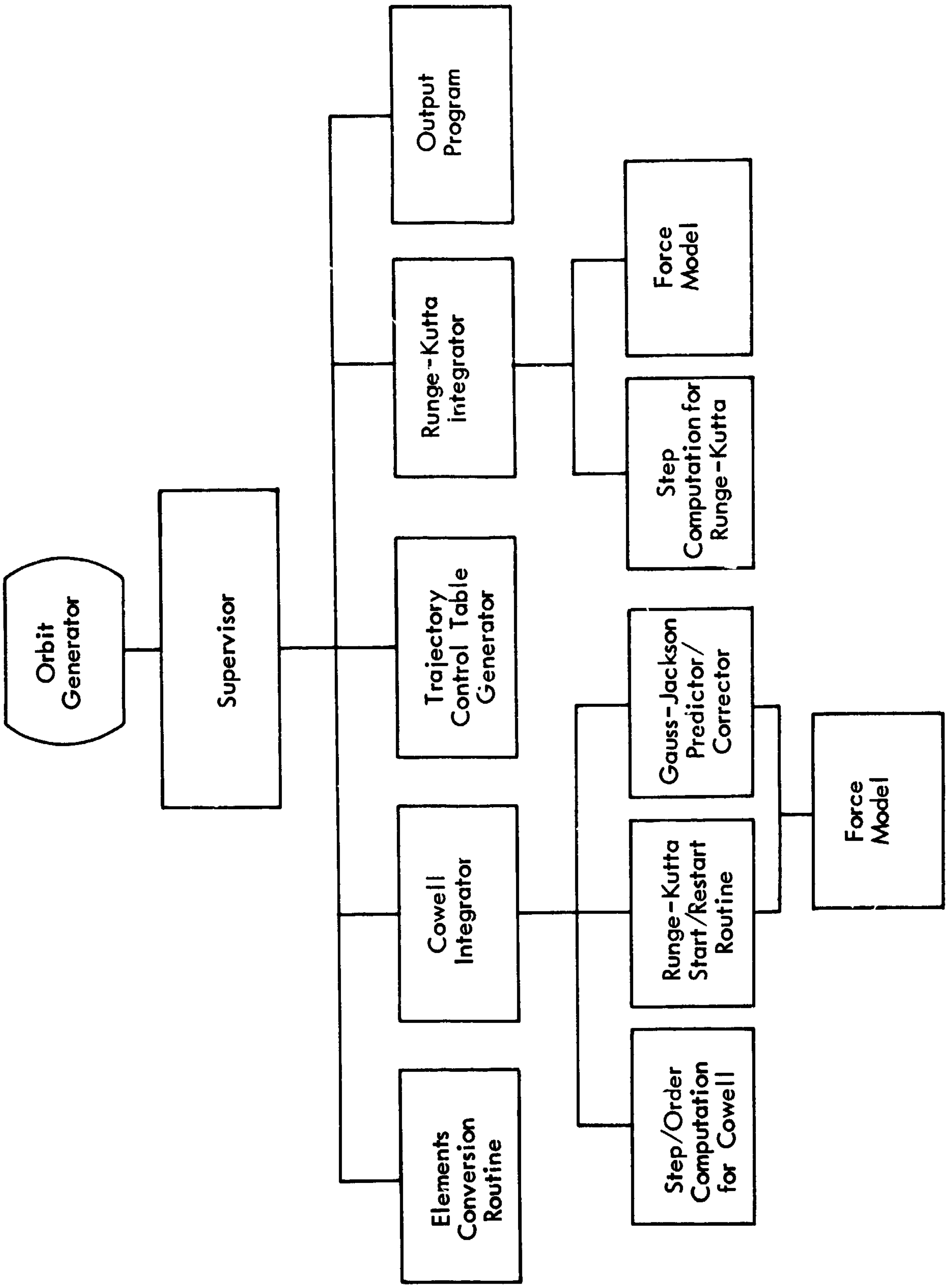


Figure 35. DODS Orbit Generator System Data Flow

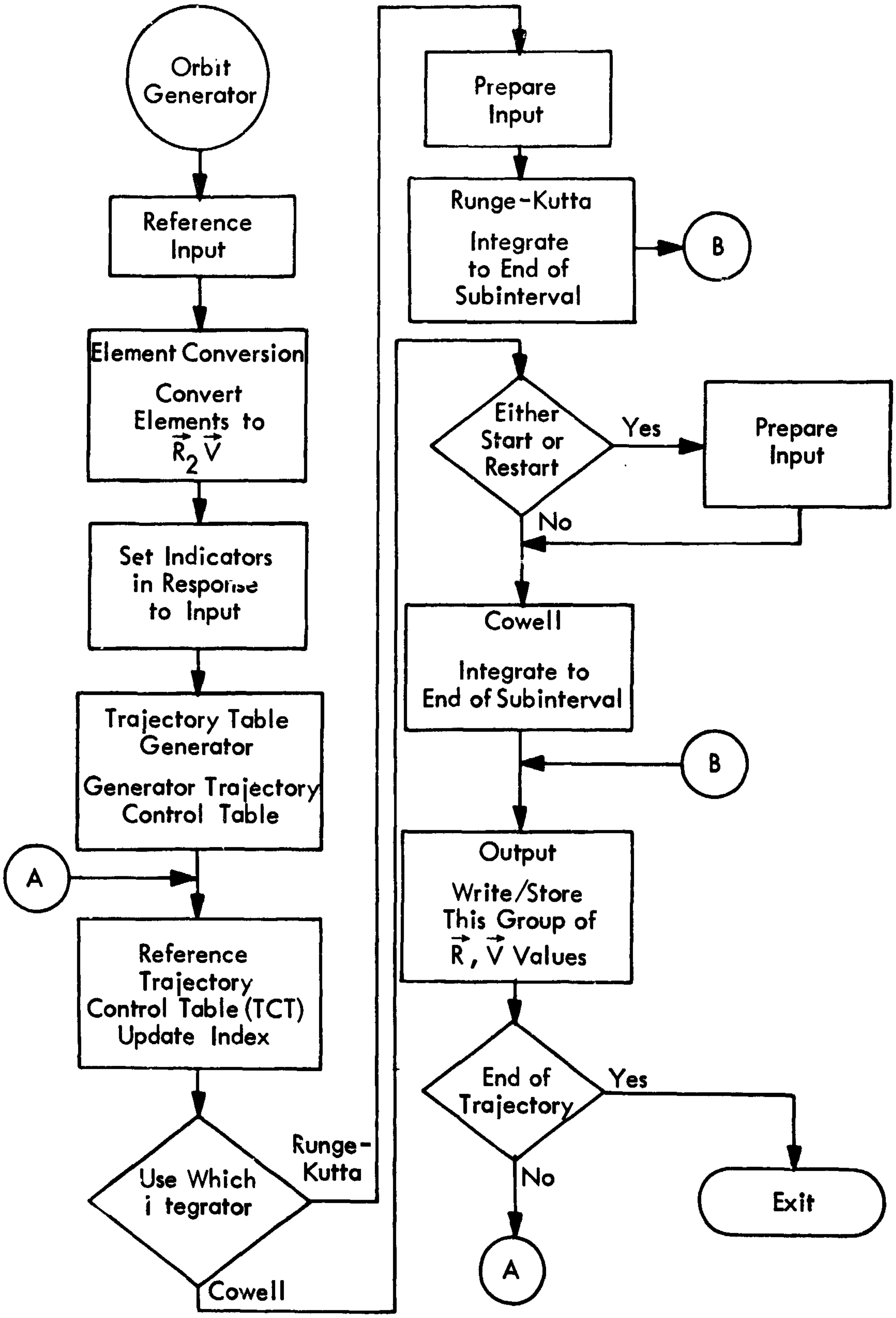


Figure 36. DODS Orbit Generator Functional Flow

flexibility that is needed in order to proceed with development in the absence of more definite specifications.

The assumptions accompanied by pertinent remarks follow:

a. There will be only one set of epoch elements, and one interval over which the orbit is to be generated.

b. The time associated with the epoch elements can fall outside of or anywhere within the interval. This includes the possibility of beginning integration during a maneuver.

c. Input to the orbit generator will always be a set of classical osculating elements ' ' epoch. Modification to handle variable input, though lacking motivatio ould not be difficult.

d. The initial direction of the integration is fixed when both forward and backward integration is to take place. This is due to ease of revision, effectively no restriction at all. However, it avoids either an additional input parameter, or additional computation.

e. Integration step size is to be less than and a fractional multiple of the output record interval for position and velocity vectors. This is consistent with the variable nature of the integration step size, and with its intrinsic lack of certainty within definable limits. In the absence of such convention, a non-trivial interpolation scheme must be employed.

f. Each maneuver is to be defined by a maneuver table. This table will contain the end points (in time) of the maneuver, and an array of velocity impulses versus time. The maneuver is effectively broken down into subintervals over which a given velocity impulse is applied. Non-alignment of subinterval points with integration points is resolved via linear interpolation.

g. A Runge-Kutta numerical integrator will be used to integrate through all maneuvers. This routine will be distinct from the modified Runge-Kutta starter used in the Cowell integrator. In addition to handling maneuvers, it will constitute a second integration technique (in addition to Cowell) in the orbit generator.

h. The step size used in the Runge-Kutta integrator will be variable but the order will be fixed. From the specifications and considerations just discussed, the following programs have evolved. They represent a solution, in discrete program form, to the problem of orbit generation in the context of the DODS.

Orbit Generator Programs

Figures 35 and 36 show the system data and functional data flow of the orbit generator. The programs are:

a. Supervisor: Maintains the first level of control in the orbit generator.

b. Elements Conversion Routine: Converts the classical osculating elements to position and velocity.

c. Trajectory Control Table Generator: Generates a table which, via its parameters, describes the entire trajectory of the satellite and from which the generation of the trajectory is controlled.

d. Cowell Integrator: Integrates free-flight motion.

e. Start Routine: Starts up integration for the Cowell integrator. It incorporates the modified Runge-Kutta technique.

f. Re-Start Routine: Re-starts integration for the Cowell integrator. This is needed when step size modification occurs. This program will either use the modified Runge-Kutta approach, or a special "difference table" modification algorithm.

g. Step/Order Computation for Cowell: Computes dynamically the integration step size and order of the difference formulas.

h. Gauss-Jackson Predictor/Corrector: Performs prediction and correction for the Cowell integrator.

i. Runge-Kutta Integrator: Integrates powered flight motion.

j. Output Program: Handles all output from the orbit generators. The output goes to direct access storage or tape. It is expected that this program will use standard I/O routines of DODS.

k. The Force Model: Although this is a major mathematical processor by itself, it is listed here because its entire utility is realized by the two integrators, above, which use it as a subprogram.

Subtask 6 - S/360 Review and Augmentation

The ultimate manner in which DODS will be implemented will be influenced to some extent by the operating system within which it must operate. This is particularly true of problem programs running in a multijobbing environment. It is therefore planned to work closely with the systems programmers at both

Goddard and IBM to evaluate the OS/360 capabilities in light of the data management, I/O processing, and user control requirements unique to DODS.

Contact has been established with the GRTS system group within IBM and systems problems common to both efforts have been discussed. It has been agreed that such items as the man/machine relationship, use of the 2250 and remote terminals, system core, disk, and drum requirements, LCS utilization, etc., will be examined in an effort to obtain effective utilization with a minimum of redundant effort. So that this be achieved it is most likely that certain system standards will have to be set and adhered to by the users of the operating system.

One of the more desirable DODS objectives is to be able to run as a background job on one Model 75 during periods when Apollo support is running. In order to ensure this, it is necessary that both DODS and the Apollo support programs be able to function under the same operating system (OS/360 Option 4). Preliminary discussions have been held with the GRTS group regarding their plans for the operating system. A preliminary version of OS/360 Option 4 has been mailed to them and they plan to begin experimentation and familiarization with the system shortly. In addition they are preparing an outline of a GRTS interface notebook, a copy of which will be sent to the DODS group. The material which is entered in this notebook will be reviewed with regard to the DODS requirements.

Subtasks 7 & 10 - Command and Control Input Formats and Manual Input Processor

A study was initiated on methods of implementing the requirments for the Manual Input Requests (MIR) and the Manual Input Processor (MIP). Preliminary discussions have led to consideration of a design which utilizes the OS/360 Job Control Language (JCL) and catalogued procedure techniques. This would involve dividing the DODS MIR's into two categories: (1) processing options which affect data sets and devices, and (2) computational options which affect the internal setting of parameter values. The processing options would be implemented as self-contained catalog procedures which would be referenced by the JCL EXEC card. The computational options would be implemented as parameter cards following a processing option.

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

Achievement of a Model 1 DODS by December 1967 is completely out of the question. Most activity on this task during the quarter has been centered around the development of the "Specified Operating Requirements" (SOR) to serve as a baseline of systems requirements from which a program system can be defined and specified for coding. To date, the SOR has not been finalized. Furthermore, activities indicate that:

a. DODS encompasses a scope far greater than originally believed

- Lunar trajectories are now to be included.
- The quantity of unknowns and elements requires derivation and coding of 5400 routines to evaluate partial derivatives.
- Correction factors for raw data and for planetary perturbations, atmosphere, etc., require acquisition of extensive sets of data and processing routines.
- A major portion of the Cowell integration system is still under analytic development and experimentation.
- The system is now to be also used for support of early orbit determination, following spacecraft launch and injection.
- Implementation of a new metric unit standards requires review of all formulation to re-insert the constant μ since it will not have a value of one in the new units.
- Inclusion of the Brouwer theory for the PCE function is now indicated.

b. GSFC needs and system boundaries are not well known. Many operational aspects of the present system should be included but these features can only be discovered and defined by a study of the usage of the present system. Furthermore, future needs are not well enough known to enable a clear statement of the system boundaries and options.

c. Due to lack of complete documentation, most communication of system uses, formulation and background is by way of meetings and personal conversations. This is increasing the time and manpower required to formulate the SOR and will similarly impact the system design.

d. A minimum of GSFC cognizant personnel have been able to devote time to the required technical meetings and conversations. To date, about 50 manhours of time on the part of GSFC personnel has been supplied where 250 manhours over the past eight weeks would have been useful.

e. Present activity is centered about finalizing the SOR and is preparatory to week no. 2 of the Design and Implementation Plan for DODS. In effect, the effort is close to six weeks behind schedule due to lack of the finalized SOR.

f. GSFC emphasis continues to center on the R & D features which understandably creates a lack of definition since only general requirements and capabilities can be described. This is inconsistent with the need for: (a) a definite SOR and specific program system design, (b) formation of the specific operational system design and in particular for detail design of the Model 1 system originally scheduled for December 1967.

Recommendations

a. It is recommended that the delivery of Model 1 be rescheduled for June 1968 to permit an orderly and thorough definition of SOR and the system design. Model 1 design should center upon definite operational capabilities, not R & D. It should include a minimum of the new mathematical formulation still under investigation.

b. GSFC technical liaison support should be increased significantly, in particular, cognizant members of programming and user groups who are familiar with the present system, during the next 10 weeks.

c. Specifically, those analytic improvements currently under development should be deferred while including in the design means for later addition of them into the system.

Task 26

ESRO II

DISCUSSION

The European Space Research Organization (ESRO), which consists of ten countries, will launch a solar astronomy and cosmic ray satellite, ESRO II, on 17 April 1967. GSFC has been requested to write an attitude determination system for this satellite as a back-up for the system being written by personnel of ESRO. This back-up system now has been coded, debugged, unit and system tested. Figure 37 is a flow chart of the system, which is being programmed for the IBM-7094. All tests were successful and the system is now ready for launch. The system consists of three parts, attitude determination, attitude prediction and control, and manual method.

Attitude Determination

The main program, ESRO, is used to determine the attitude by finding the intersection of a number of space cones. This method is based on the idea that from a variety of sensor data which relates to angles between the spin axis and known spatial directions, such as sun vector and magnetic field vector, the spin axis is constrained to lie simultaneously on a number (n) of space cones having known unit vectors $\overline{V}_i$ as axes and known angles 0_i as generating angles ($i = 1, 2, \ldots, n$). Thus by finding the intersection of two or more cones it is possible to ascertain the right ascension and declination of the spin axis. This is done by finding the intersection in the sense of weighted least squares by a differential correction approach starting from an a priori attitude. Details of this procedure, together with associate error analysis, are given under subroutine CONES in the Attitude Determination System for AE-B Satellite (Ref 7). Included as part of this program is a subroutine, PESROD, which first calls on subroutine

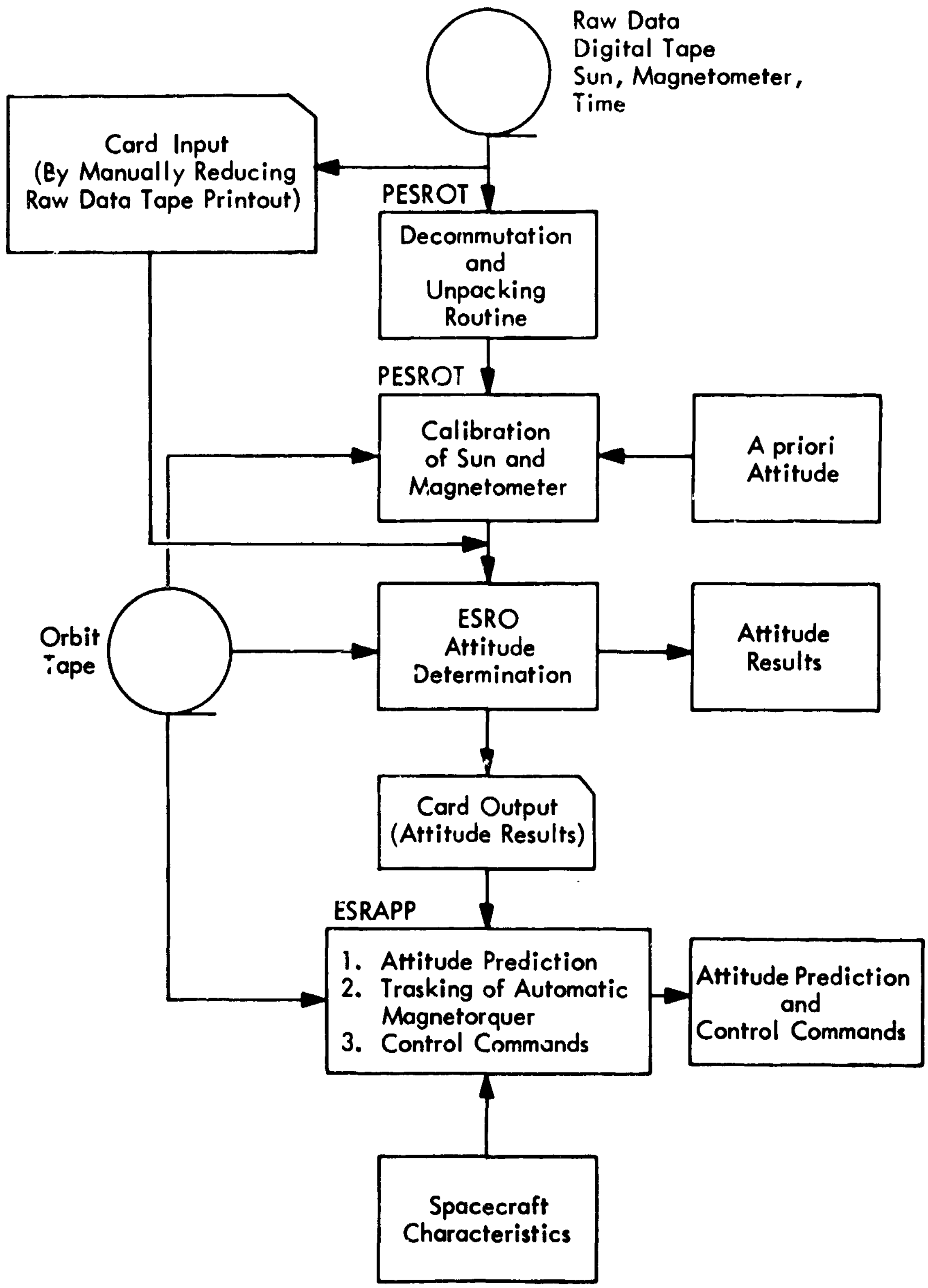


Figure 37. ESRO II Attitude Determination System Data Flow

PESROT to read and unpack the data tape. The data returned are triads of magnetometer readings (x, y, and z), pairs of sun angles, and associated times. Also, bits are set to note when the magnetorquer is on. PESROD converts the times into minutes from DREF and converts both the sun and magnetometer PCM (pulse coded modulation) counts from the tape into degrees and milligauss, respectively, using linear functions. Then the magnetometer data is screened so that it is used only when the magnetorquer is off, when sense switch six is up; when sense switch six is down, there is no screening of the data. Next the sun data is processed. At each sample time there are two readings—one from each sensor. If the two readings are not within one degree of each other, both are ignored. The sun data is then averaged over either 20 minutes or until the sun angle (α) has changed two degrees. Finally, the sun angles and magnetometer readings are returned to the main program.

Attitude Prediction and Spacecraft Control

An attitude prediction program, ESRAPP, is used to predict the attitude, steer the spacecraft, and track the automatic magnetorquer. ESRAPP will predict the attitude of the ESRO II satellite whether the craft is operating under the automatic magnetorquing system or under the manual override. Due to the characteristics of the ESRO II spacecraft it was found that TIROS MGAP could be used, with some additional logic, to track the satellite. This logic tracks the setting of the magnetorquer coil which operates much like the QOMAC coil in TIROS, with the exception that the magnetorquing coil can be initiated automatically from within the spacecraft. Whereas, the QOMAC coil in TIROS operates in two cycles/orbit. That is, the coil is positive and negative twice per orbit. The QOMAC coil starts with a positive sign and then changes to negative after a quarter orbit and then positive for the next quarter orbit, and negative for the last quarter orbit. In this way the satellite retains the proper attitude.

All the logic for TIROS MGAP has been retained in ESRAPP with some additions being made in TORQUE (a subroutine of MGAP). The additions disable the QOMAC cycle and enable the automatic magnetorquing process.

ESRAPP predicts automatic and manual switch settings of the magnetorquer. Under automatic control the magnetorquer is enabled when $80^{\circ} \geq \alpha \geq 100^{\circ}$.

The magnetorquer coil is turned on when the strength of the earth's magnetic field exceeds a threshold of 100 milligauss, as opposed to QOMAC which is turned on for an entire orbit. This action of the magnetorquer applies a torque to the spin axis to force gamma to 90^o. At this point the coil is turned off and the automatic system is disabled until such time that the gamma angle again exceeds the above values.

Under manual control, magnetorquer coil settings are input and the automatic logic is bypassed. In this mode of operation, ESRAPP predicts when the magnetorquer is to be turned on and off. Various other options have been added to the ESRAPP routine. These include variable input time intervals and an extensive debug system via sense switches 5 and 6.

Numerous tests have been performed on the ESRAPP routine. Among these are a long range (90 days) drift study, a smaller integration period (.05 min), long range steering using the automatic and manual systems, testing for a gamma of over 100^o and under 80^o, and testing for a variable magnetic field threshold. All tests were successful.

Manual Method

This technique permits the manual search of a computer printout of the raw digital tape for sufficient sun and magnetometer data, each with the time of observation, for input to the attitude determination program ESRO. This method was devised as an emergency procedure.

PROGRAM FOR NEXT REPORTING PERIOD

To prepare for the launch, scheduled for 17 April 1967, and to make any changes to the system that may be required because of hardware changes or unexpected contingencies. A two-week support task will commence on the day of the launch.

CONCLUSIONS AND RECOMMENDATIONS

The ESRO II attitude determination system is now operational and ready for launch. It is recommended that as soon as all the spacecraft elements are available, another system test be conducted with these elements.

Task 29

ATS-B PFM PROGRAMMING SUPPORT

DISCUSSION

The ATS-B PFM Programming Support task is a new task which was approved 30 March 1967. Initial analysis began on redesign of the ATS-B, PHASE0, COMVER, Sequence Count Print, and Buffer Tape Print programs for the UNIVAC 1108.

PROGRAM FOR NEXT REPORTING INTERVAL

The analysis, coding, and unit testing of the PHASE0, COMVER, Sequence Count Print and Buffer-Tape Print Programs are expected to be completed. Also scheduled for completion are the analysis on the redesign of the ATSBMN main program and writing and publication of the specifications for the new ATSBMN program.

Section III

NEW TECHNOLOGY

This section describes items of new technology which have been selected from work performed during this quarter. These potential new technology items are being submitted formally for acceptance via prescribed procedures.

One item from last quarter (Computerized Teletype Data Message Scheduler for Satellite Launch Support) has been withdrawn. This item was not of sufficient scope to be regarded as new technology.

ABSTRACT

Determination of Magnetometer Mounting Angles

Problem

Part of the hardware designed into the Atmospheric Explorer-B (AE-B) spacecraft was an orthogonal triad of magnetometers mounted with one magnetometer of the triad in alignment with the satellite spin axis. This configuration was intended to provide an important parameter for attitude determination. It became apparent after launch, however, that the triad was neither orthogonal nor mounted properly with relation to the spin axis. As a result, analyses were performed to determine the magnetometer mounting angles and equations were developed to utilize the data from the non-orthogonal system. The analyses and formulas are derived in the following description.

New Concept

This concept describes a process for evaluating sensor mounting angles for a non-orthogonal triad of magnetometers on a spin-stabilized satellite and a method of determining the angle between the satellite spin vector and the geomagnetic field vector using the magnetometer readings.

Technical Merits

Most magnetometer systems are designed as orthogonal triads for simplicity in computations. The development in this paper gives the equations which must be used with a non-orthogonal system.

Other Applications

The equations are applicable to attitude determination computations for any spin-stabilized satellite using a magnetometer triad.

ABSTRACT

Development of a One-Pass Telemetry Data Reduction System for the ATS-B Satellite.

Problem

Existing satellite data processing systems require interruptions of computer processing for analysis of time continuity. These interruptions may result in a shipment delay of up to five days before final decommutated tapes can be sent to experimenters, even though 90 percent of the data is often perfect. Moreover, the human application of corrections is subject to error, because of the extremely large volumes of data being processed by these satellite systems.

New Concept

Computer processing interruptions may be eliminated by the implementation of a one-pass data reduction system which utilizes the concept of an automatic time analysis and correction procedure as described in the IBM publication "A Generalized Approach to Correction of Time Errors of Spacecraft Telemetry" by Mr. R. R. Rosner.

Technical Merits

The ATS-B system stores all the data on a disk and writes only one set of tapes for record retention. Thus error prone tape handling of data is reduced. The time correction algorithm is faster, more accurate and more consistent than present methods. The accuracy and consistency are derived from a history that is kept of every file processed. Previous manual methods compared only those files currently being processed.

Other Applications

The algorithm described by Mr. Rosner was designed for the Applications Technology Satellite (ATS) series; however, the generality of the time correction procedure allows its adaptation to similar telemetry systems.

ABSTRACT

Positioning a Magnetic Tape Data Set Opened for QSAM Processing with OS/360

Problem

When reading magnetic tape data sets (files) it is often desirable to start processing at a record other than the first one in the data set. In many cases the number of records to be spaced over is in the hundreds and sometimes in the thousands. The System/360 QSAM offers no facility for spacing over records within a data set.

New Concept

Magnetic tape data sets opened for processing with QSAM can be positioned to a desired location within the data set using the EXCP macro and an appropriate series of channel command words (CCW's).

Technical Merits

The advantage of this technique is that tape movement is continuous and only one START I/O is executed for one CCW series. Thus, actual tape movement is optimized and the total time required both of the channel and of the supervisor is kept to a minimum.

Other Applications

This concept applies to any user who selects data sets under QSAM of OS/360.

Section IV

GLOSSARY

ADS	Attitude Determination System
AE-B	Atmospheric Explorer Satellite B
AEPROC	Data processing program for AE-B
AEB-MGAP	Magnetic attitude prediction program for AE-B
ALPHA	Right ascension
ASP-MGAP	Attitude prediction program
ATS	Applications Technology Satellite
ATSOCC	Applications Technology Satellite Operation Control Center
AUTODOC	Automatic documentation
AVCS	Advanced Vidicon Camera System
BBRC	Ball Brothers Research Corporation
BIOS	Biosatellite
BLUEBIRD	One of the COMSAT satellites to be launched
BSAM	Basic Sequential Access Method
CDUST	Core Duplicating System Tape
COMSAT	Communications Satellite Corporation
COMPARE	Compare two trajectories
CONES	AE-B subroutine for attitude determination
CONTDC	Continue DC
CORFOE	Program to compute orbital elements
DATPRO	TOS/ESSA attitude determination program
db	Decibels
DC	Differential correction
DCS	Direct Couple System
DECOM	Decommutation routine

DELTA	Declination
DHE	Data Handling Equipment
DODS	Definitive Orbit Determination System
DREF	Day of reference
DSAI	Digital Solar Aspect Indicator
DTO	Detailed Test Objectives
EFM	Element File Maintenance
EGO	Eccentric (Orbit) Geophysical Observatory
ELK	Electron Shower Calculation
EME	Environmental Measurement Experiment
Encyclopedia Tape	Records of detailed scientific satellite experiment data
EPE	Energetic Particles Satellite
EPHEM	Satellite Ephemeris Generator
ESRO	European Space Research Organization
ESRO II	European Space Research Organization satellite
ESRAPP	FORTRAN II subroutine to predict attitude for the ESRO II satellite
ESSA	Environmental Science Services Administration
FORTRAN II AND IV	Formula Translation - the language used for scientific computer programming
FUSIT	Hughes Aircraft Company programs for determining time and location for igniting apogee motors
GAMMA	Sun angle
GAMWIN	Program to compute launch window
GMT	Greenwich Mean Time (Universal Standard Time) Time at the zero meridian, Greenwich, England; also Zulu or UT
GREMEX	Goddard Research Engineering Management Exercise
GSDAS	Generalized Satellite Data Analysis System
GSFC	Goddard Space Flight Center
HAC	Hughes Aircraft Company
H_2O_2	Hydrogen peroxide
IMP	Interplanetary Monitoring Platform
INJUN	Rocket booster for small, non-orbiting payloads

IR	Infrared (sensor)
JCAL	Julian calendar program
JCE	Jet Control Electronics
JIFFY	Program to smooth MGAP results
KP	Geomagnetic activity indexes
Launch Window	Period of time in which favorable conditions exist for launching a satellite
LCS	Large Core Storage
LNRORB	Lunar orbit program
Logbook Tape	Summary records of scientific instrument data
LOS	Line of sight
MAC	Magnetic Attitude Control
Magnetorquer Coil	Coil on board the ESRO II satellite that produces a torque to force the gamma angle to return to the desired position
MAS	Multi-Application Subroutines
MCC	Master Control Console
MGAP	Magnetic attitude predictor
MLT	Magnetic Local Time
MIP	Manual Input Processor
MIR	Manual Input Requests
MODULE	A functionally related set of instructions and data applicable to DODS effort (see Task 25)
O1MERG	ORB1 merge program
OA	Optical Aspect
OFM	Observation File Maintenance
O-C	Observed minus computed
ORB1	Orbital position tape; contains satellite position vectors and velocity vectors at discrete intervals
OS/360	Operating System/360
OSO	Orbiting Solar Observatory
OSOCC	OSO Control Center
OWSP	One Word Storage Programmer
P	Elliptical orbit period

P_e	Earth sidereal period of angular rotation
PACE	Phased Array Control Antenna
PCM	Pulse Code Modulation
PDD	Module performance and design description which is the documentation format for the completed DODS modules
PDS	Module performance and design specification which is the documentation format for the DODS design effort
PESROD	FORTRAN IV subroutine to calibrate and screen data for ESRO II satellite
PESROT	MAP subroutine to extract data for ESRO II satellite
PFM	Pulse Frequency Modulation
POGO	Polar Orbiting Geophysical Observatory
POLANG	RF Signal Polarization Angle
PREPRO	Processing program for TOS/ESSA
PUP	Prelaunch and Utility Programs
QOMAC	Quarter Orbit Magnetic Attitude Control - coil on board the TIROS satellite that produces a torque to force the sun angle to return to the desired attitude
QPR	Quarterly Progress Report
QSAM	Queued Sequential Access Method
RAE	Radio Astronomy Explorer
RES	Roll equation smooth
SFL	Southern Frame Limit
SCO	Subcarrier oscillator (real-time)
SDA	Scientific Data Analysis
SIN	Sine
SLP	System Log Processor
SOR	Specified Operational Requirements
SORE	SORE Count
STADAN	Satellite Tracking and Data-Acquisition Network
STADEE	Status Data Extraction Evaluation and Reduction program
STARSII	Satellite Telemetry Automatic Reduction System II
TAPRE	Program to read ORB1 tapes
TEC/TTCC	TOS Evaluation Center/TIROS Technical Control Center

TIMSPN	AE-B subroutine to compute the spin rate using mid-scan times
TIROS	Television Infrared Observatory Satellite
TIROS MGAP	FORTRAN II subroutine to predict attitude for the TIROS satellite
TOS	TIROS Operational System
TORQUE	FORTRAN II subroutine of ESRAPP
V	Orbit velocity
VCO	Voltage controlled oscillator
W_e	Earth sidereal rate of angular rotation
WMSAD	World Map and Station Acquisition Data program
WMSUM	World Map Summary program

Section V

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Appendix A

ELECTRON SHOWER CALCULATION

The electron shower calculation (ELK) program calculates the losses suffered by electrons or photon (gamma ray) upon passing through a multi-layer absorber.

Losses occur in several ways:

a. Electrons may interact to produce secondary electrons and secondary photons.

b. Gamma rays may interact with the material to produce secondary electrons and photons.

c. Secondary particles of high energy may interact as in (a) and (b).

d. Compton Scattering may produce photons.

A random number generator is used to generate particles (mathematically) in proportion to their probability of occurrence. Interactions are generated in proportion to their interaction probability.

Losses are thus calculated as a function of depth in the absorber and numbers of each kind of particle.